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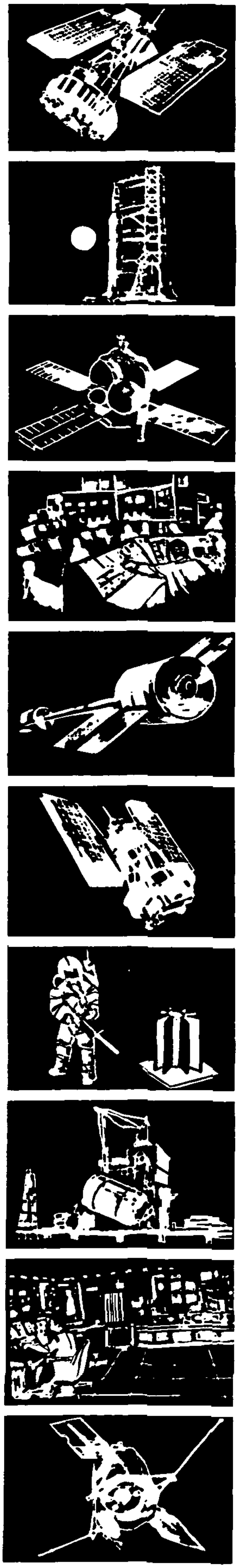

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EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY

Report No. 7

EOS SYSTEM DEFINITION REPORT

GENERAL ELECTRIC

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Report No. 7

EOS SYSTEM DEFINITION REPORT

Prepared for:
GODDARD SPACE FLIGHT CENTER
Greenbelt, Maryland 20771

Under
Contract No. NAS 5-20518

GENERAL ELECTRIC
SPACE DIVISION
Valley Forge Space Center
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SECTION 1.0

INTRODUCTION

This report, "EOS System Definition," has been prepared for NASA/GSFC under Contract NAS5-20518, EOS System Definition Study. The report is a concise summary of all the work performed during the contract and incorporates NASA tradeoff decisions, payload changes, and other factors which affected the course of the study during the period of performance. General Electric's preferred design approach for the EOS program is presented.

The EOS System Definition Study resulted in a modular design of a General Purpose Spacecraft, a Mission Peculiar Spacecraft Segment which performs the EOS-A mission, an Operations Control Center, a Data Processing Facility, and a design for Low Cost Readout Stations. The study verified the practicality and feasibility of the concept of a low-cost modularized spacecraft having the capability of supporting many missions in the Earth Observation spectrum over the next ten years. The applicability of the shuttle system for retrieval and service of the spacecraft was also defined and verified during the study.

The Definition Study Schedule is shown in Figure 1-1. The schedule indicates the seven reports that were required for the study and their delivery dates, as well as the dates for the oral presentations summarizing these reports. The significant study results are documented in the first six reports, all of which are summarized in this, the seventh report.

This final report and study reports No. 5 and 6 incorporate the major EOS-A requirement/guideline changes made by NASA during the course of the study. The most significant change was the payload which originally was a Thematic Mapper (TM) and a High Resolution Pointable Imager (HRPI) on an R&D oriented mission. The Payload was changed to a 5-Band Multispectral Scanner and a Thematic Mapper with the mission considered to be operational for the MSS and the TM as an R&D "piggy-back" experiment.

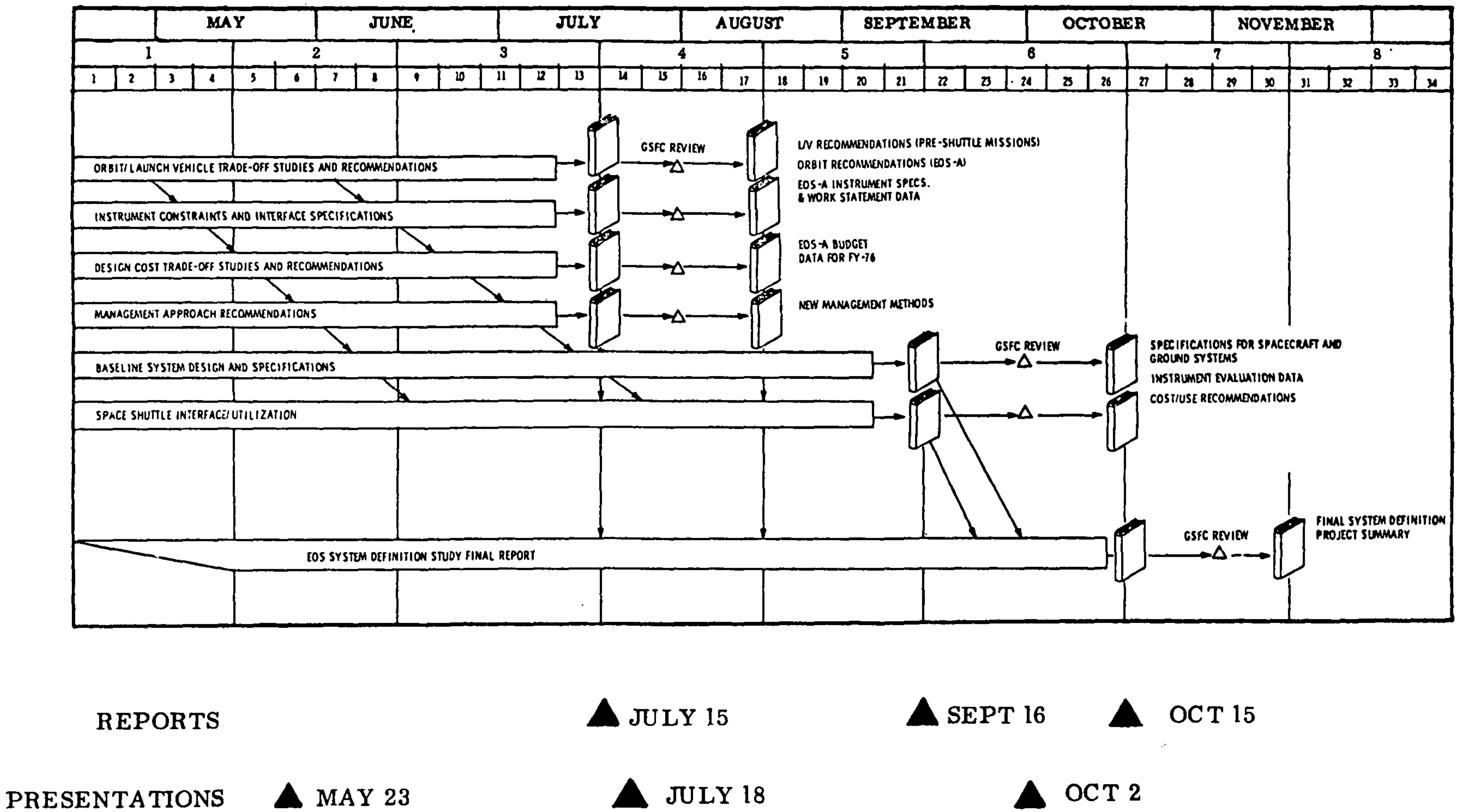


Figure 1-1. Study Schedule

During the study, special emphasis was placed upon designing a low-cost, modular, flexible, standard multi-mission spacecraft (described in Report No. 5). The study, therefore, culminated in descriptions and specifications for the general purpose spacecraft, which could support any number of missions, plus the EOS-A mission peculiar spacecraft and ground segments.

The Data Processing Facility is designed to support the EOS-B instruments consisting of a Thematic Mapper and a High Resolution Pointable Imager. The Operations Control Center is designed to support the general purpose spacecraft with software changes required to support the mission peculiar aspects of different payloads. A Low Cost Readout Station has been designed to receive selected (reduced bandwidth) Thematic Mapper or MSS data.

The organization of this final report is based on the six major reports issued during the study. Each major section of this report summarizes the significant results of each of the six study reports as shown below:

This Report	Summarizes	
Section 2	Report No. 1	Orbit/Launch Vehicle Tradeoff Studies and Recommendations
Section 3	Report No. 2	Instrument Constraints and Interfaces
Section 4	Report No. 3	Design/Cost Tradeoff Studies and Recommendations
Section 5	Report No. 4	Low Cost Management Approach and Recommendations
Section 6	Report No. 5	Baseline System Design and Specifications
Section 7	Report No. 6	Space Shuttle Interface/Utilization

SECTION 2.0

ORBIT/LAUNCH VEHICLE TRADEOFF STUDIES AND RECOMMENDATIONS

This section presents a summary of the driving constraints/requirements on the EOS-A orbit/launch vehicle selection, summarizes the propulsion system (Hydrazine) and launch vehicle (Delta 2910) selected for EOS-A and presents the rationale for selection of the recommended EOS-A orbit (775 km). The choice of the recommended orbit for EOS-A, whose payload includes a Multispectral Scanner (MSS) and a Thematic Mapper (TM), was impacted by the later EOS missions such as EOS-B where the payload contains a High Resolution Pointable Imager (HRPI). Thus the orbit selected was a tradeoff between the existing Multispectral Scanner (MSS) orbit and the orbit preferred for HRPI/TM.

2.1 REQUIREMENTS/CONSTRAINTS/CRITERIA

The driving constraints for the orbit/launch vehicle selection can be separated into three key areas:

- mission constraints
- launch system constraints
- spacecraft weight constraints.

The MISSION CONSTRAINTS that drive the orbit altitude selection can be summarized as:

- real-time coverage of USA
- sidelap width . . . . 5 to 15% (prefer 10% maximum)
- repeat cycle . . . . 15 to 18 days
- access time (HRPI) . . . . 2 to 4 days
- maximum offset pointing (HRPI) . . . . 45^{o}.

Five candidate orbits that meet the majority of these mission constraints have been identified. The altitudes of these candidate orbits range from 665 km to 790 km. The lower limit was selected as the practical minimum altitude due to drag effects while the upper altitude limit was set by the launch system performance, including the ability to directly access the orbit with shuttle.

The LAUNCH SYSTEM CONSTRAINTS for the primary launch vehicles are shown in Table 2-1. Titan IIID NUS has been eliminated due to the high launch cost of between 25-44M dollars.

The launch system costs (which include the on-board propulsion system) vary from $6.6 to $12.9M, while the allowable spacecraft weight varies from 2380 to 4520 pounds, depending on launch vehicle and mission altitude. The Titan shroud volume is approximately three times the volume of the Delta shroud.

Table 2-1. Launch System Cost/Weight/Volume Comparison

Launch Vehicle	Launch Vehicle Cost ('74 Dollars) M	Prop Syst Cost M	Total Cost M	Shroud Volume FT^3	Allowable S/C * Weight (lbs) (Minus Propulsion)	
					650 Km	740 Km
Delta 2910	6.0	0.6	6.6	600	2520	2380
Delta 3910	8.0	0.6	8.6	600	3520	3340
Titan IIIB NUS	12.2	0.7	12.9	1670	4520	4340

* Allowable weights assume spacecraft returned to shuttle for retrieval.

The SPACECRAFT WEIGHTS for alternate Delta and Titan spacecraft are shown in Table 2-2.

Table 2-2. EOS Spacecraft Weights

	Delta Spacecraft Wts (lbs)				Titan Spacecraft Wt (lbs)	
	EOS-A	EOS-A'	Combined S/C	EOS-B	Combined S/C	EOS-B
Basic S/C	1075	1075	1075	1075	1164	1164
Mission Peculiar	572	479	605	565	605	565
Payload	505	155	660	700	660	700
Total Minus Propulsion	2152	1709	2340	2340	2369	2429

2.2 MISSION ORBIT ANALYSIS

There are a multitude of orbits which potentially meet the EOS mission requirements. They range from the ERTS-type (18 day repeat cycle, daily progression of the ground track) which are adequate for the Thematic Mapper, to various types of interlaced orbits which provide shorter access time to points on the Earth when using a HRPI instrument. The parameters of interest when selecting the EOS orbit are given in Table 2-3 along with some discussion as to how each of these parameters affect the mission and system. The values selected for each of the parameters are also identified.

During the course of the study, all of the potential orbits were investigated considering both the EOS-A (MSS and TM) and EOS-B (TM and HRPI) missions. The tradeoff studies concluded that the most cost-effective approach is to select an orbit based on the EOS-B mission such that it is optimized for both the TM and HRPI instruments and modify the EOS-A MSS to be compatible with the selected orbit. No changes are then required to the TM design for future missions as it progresses from an R&D instrument to a fully operational one.

Additional key constraints considered during the orbit selection analysis are

- direct shuttle access
- minimization of orbit adjustments due to atmospheric drag effects
- Delta and Titan Launch capability
- Ground station coverage and contact times.

The 775 km orbit was selected since it is the best overall compromise of the above constraints, provides 3-day HRPI access and is fully compatible with TM designs.

A typical one-day ground coverage trace is shown in Figure 2-1 for the daylight portion of each orbital revolution. The day-to-day pattern is shown in Figure 2-2. The two outer orbits (Orbit 1 Day 1, Orbit 2 Day 1) represents the ground traces of adjacent orbits on a single day. On the second day a ground trace (Orbit 1 Day 2) falls approximately one-third the way between the two. On the third day, a ground trace (Orbit 1 Day 3) falls two-thirds of the way between the first two. The HRPI off-nadir pointing

Table 2-3. Orbit Tradeoff Parameters

PARAMETERS	DEFINITION	SIGNIFICANCE FOR EOS-A	SELECTED VALUE
Coverage	The amount of Earth which can be imaged by the satellite instruments.	Sun-synchronous orbits provide the opportunity for the satellite to view the entire earth up to approximately 81° latitude. Amount of coverage is then determined by the swath width.	Global
Swath Width	The width of sensor field of view as projected on the ground.	Determines the width of the imaged data and is a key trade-off parameter in sizing the instrument optics and detectors.	185 km
Sidelap	The overlap between adjacent swaths as projected on the ground. Normally expressed as a percentage of the swath width and measured sidelap at the equator; sidelap increases with latitude.	Sufficient sidelap insures that no Earth surface area is lost between the swaths. ACS tolerance & orbit control precision will cause actual sidelap to vary about the nominal value. The more accurate the ACS & orbit control, the smaller the sidelap requirement. Excessive sidelap, while insuring coverage creates additional processing load on the ground system. ERTS had 14% sidelap. EOS ACS and orbit control performance can easily accommodate 5% sidelap.	11% at equator
Sun Synchronous	An orbit in which the orbital plane rotates at the same rate as the mean rate of the Earth about the sun.	Any point on the Earth will always be imaged at the same mean sun time which minimizes illumination changes on the ground scene. Also simplifies the power subsystem design by permitting one axis drive on the solar array & the thermal & ACS designs by limiting the variation in sun angle.	Sun Synchronous
Descending Node Time	The mean solar time the satellite crosses the equator on the North to South portion of its orbit. This terminology used primarily with sun synchronous orbits.	Affects Beta angle variation which in turn affects both thermal & power subsystem design. From user viewpoint, affects illumination conditions of the scene. Can be selected independently of all parameters.	1130
Repeat Cycle	The time required for the satellite to (almost) exactly repeat its coverage pattern.	Provides the opportunity to gather data periodically under relatively similar conditions. Simplifies flight operations and data processing. Users generally prefer shorter repeat cycles.	17 days
Access Time	The minimum time between two sequential observations of the same point on the ground; it need not be under the same relative conditions. Sometimes defined as the time between an event occurence and when it can be observed.	Dictates HRPI offset pointing angle since offset pointing provides the opportunity for reduced access time. (Note in an ERTS-type orbit, repeat cycle equals access time.)	3 days
Interlace Pattern	The pattern in which successive ground tracks "fill in" between two adjacent orbits on a given day during one full repeat cycle.	Determines the time between the imaging of adjacent ground swaths. For a given access time this time can range from the access time up to one half the repeat cycle. The longer the time between adjacent swaths, the more the scene characteristics can change hence the poorer the mosaicing potential.	Equal to the access time
Altitude	The mean distance between the satellite orbit and the Earth radius.	Determines repeat cycle & interlace pattern. Affects instrument optics and detector sizing for a given coverage & swath width. Also affects launch vehicle payload capacity integral propulsion requirements and Shuttle service/recovery capability.	775 km

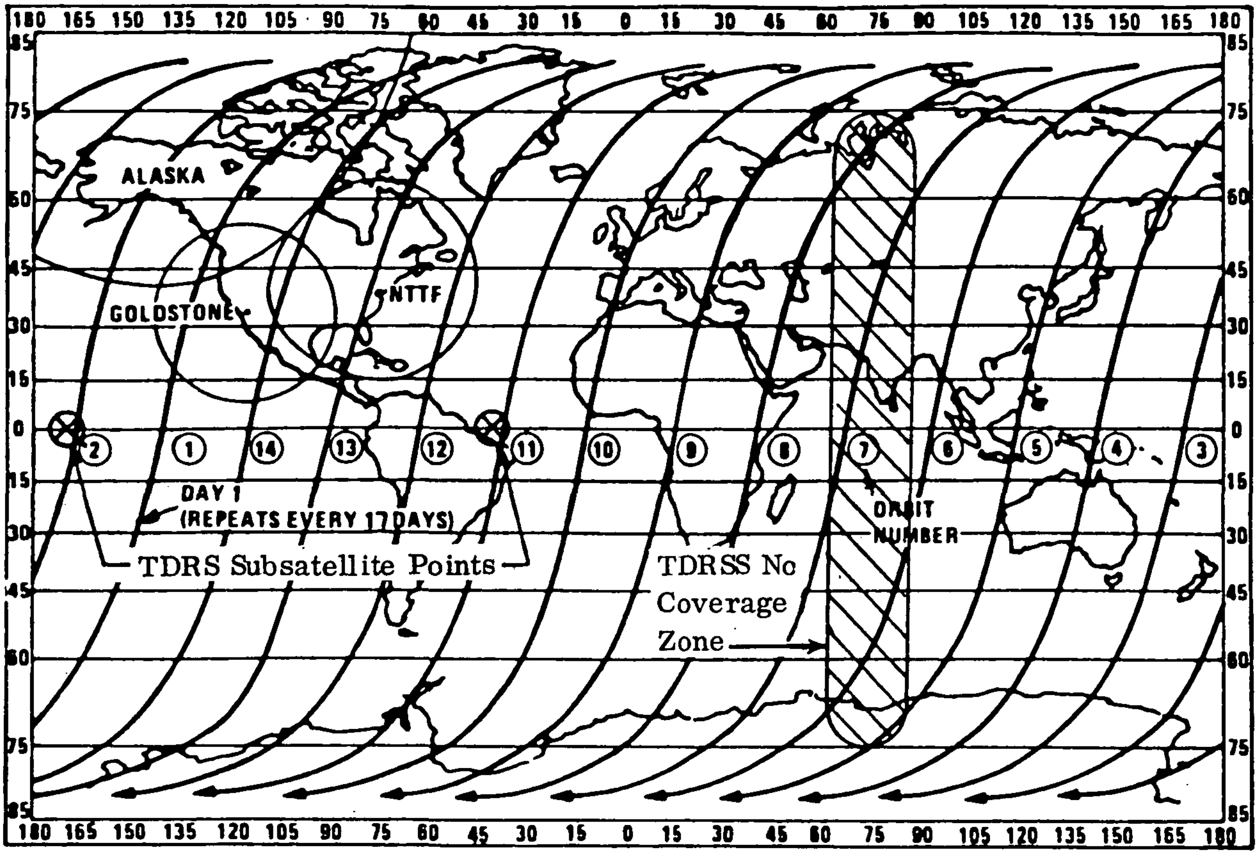


Figure 2-1. Typical EOS-A Daily Ground Trace (Daylight Passes Only)

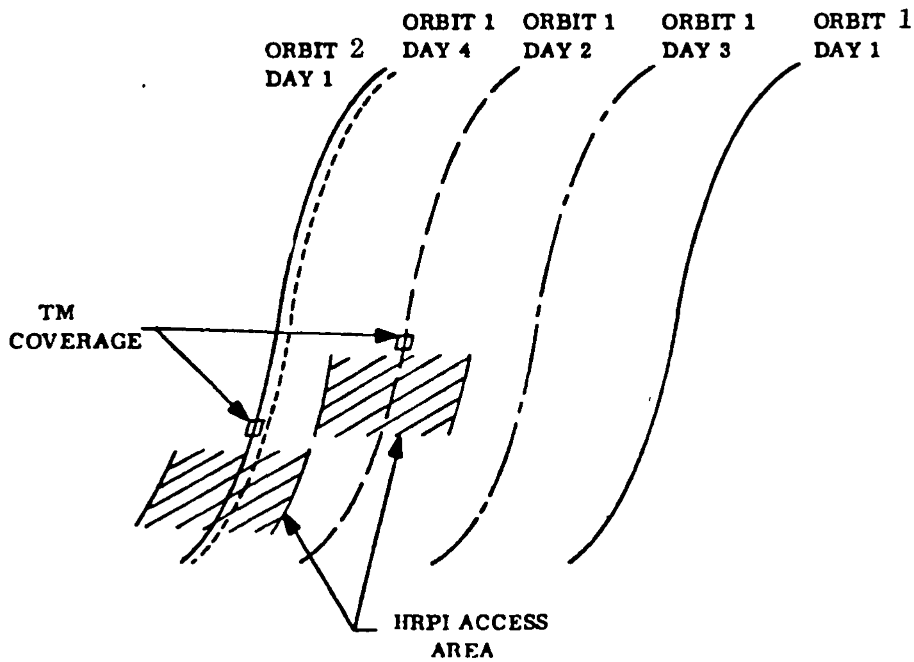


Figure 2-2. EOS-A Orbit 3-Day HRPI Access

capability is equal to one-third the distance between the two swaths on the first day; hence potential access anywhere on the earth is provided every three days without requiring more than a 32^{o} offset view from nadir.

2.3 SPACECRAFT PARAMETRIC PERFORMANCE ANALYSIS

Orbit altitude effects on spacecraft and payload design has been evaluated over an altitude range of 520 to 1660 km.

/

Specific spacecraft performance/cost/trades were made for:

- Power subsystem
- ACS subsystem
- Wideband/C&DH subsystem
- Thermal subsystem
- Propulsion subsystem.

It is significant that no "driving" cost factors were identified for any of the spacecraft subsystems that would constrain orbit altitude selection.

Payload performance trades were done on a relative basis using the Thematic Mapper as a representative instrument. Theoretical cost and weight versus altitude curves for instruments show a direct relationship between orbit altitude and weight and cost. In practice instrument designs are expected to be based on a few discrete aperture sizes. For a given aperture size, performance would be allowed to vary over small altitude ranges. Stepped cost curves thus result over a large altitude range wherever the aperture size changes. The instrument costs are therefore expected to remain relatively constant over the small range of altitudes under consideration for EOS-A.

The MSS is presently designed to provide 80 meter IFOV and 15 Mbps data rate from 914 km. The recommended redesign for lower altitude application of the MSS is to increase the IFOV and maintain the 80 meter resolution and the data rate. The tradeoff studies indicate that the cost effective approach to increasing the IFOV is to change the optics.

2.4 LAUNCH SYSTEM PERFORMANCE ANALYSIS

The EOS-A launch system includes the launch vehicle, integral propulsion system and shuttle retrieval system. The integral propulsion system is included since it plays a major role in achieving the mission orbit when Titan launch vehicles are considered and also provides the capability of retrieving the spacecraft by shuttle at an altitude other than the mission altitude. Launch system performance for Delta 2910, Delta 3910, and Titan IIIB is summarized in Figure 2-3.

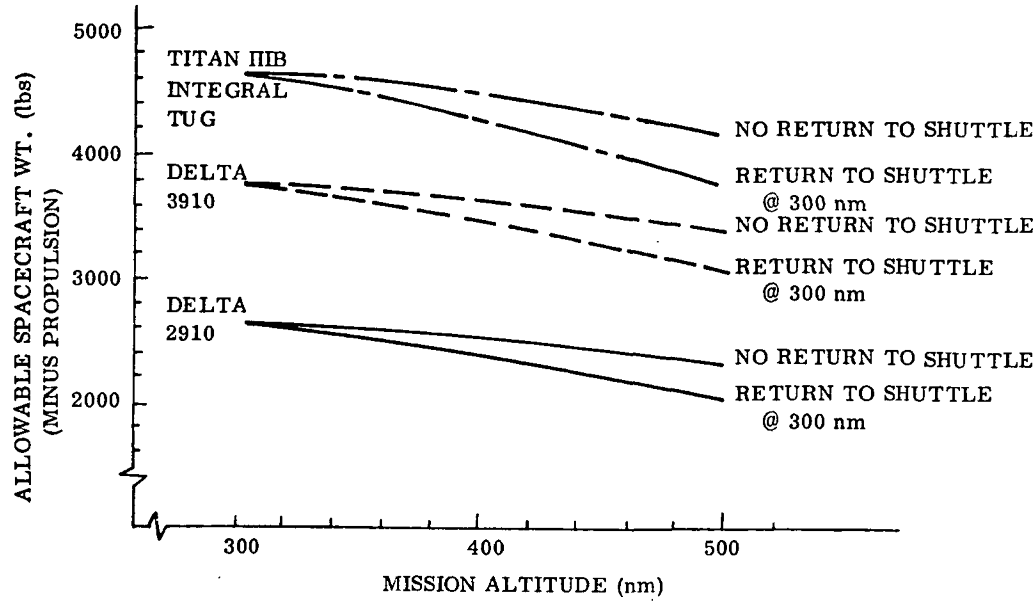


Figure 2-3. EOS-A Launch Vehicle Performance (Hydrazine Propulsion System)

2.5 ORBIT/LAUNCH VEHICLE SELECTION

The selection of the preferred orbit and launch system for EOS-A involves trade studies between the type of propulsion system, launch vehicle and orbit altitude while also impacting a wide range of variables. To simplify the selection process, preliminary choices were made for the propulsion system and launch vehicle and the impacts of alternate mission altitudes determined to establish the preferred mission altitude. It was then necessary to evaluate if any alternate selection of propulsion system or launch vehicle would impact the selection of mission orbit.

The Hydrazine propulsion system was selected due to its low cost for EOS-A in addition to its flexibility and low cost in meeting the requirements of the total mission model. The all hydrazine system proved lowest cost for Delta, Titan, or Shuttle applications.

The Delta 2910 was selected as the preferred launch vehicle for EOS-A since it is the lowest cost launch vehicle that can perform a meaningful EOS-A mission. The Delta 2910 can launch the Delta spacecraft to 775 km with sufficient spacecraft propulsion on-board to return the spacecraft to 611 km for Shuttle retrieval. The 775 km altitude is also directly Shuttle accessible (with a cost penalty) should recovery be required without use of the on-board propulsion system.

The mission orbit of 775 km was selected as the best compromise between Shuttle compatibility, mission compatibility, ground system compatibility, launch system impacts, spacecraft impacts, payload instrument impacts, and impacts of later missions as summarized in Table 2-4.

Table 2-4. EOS-A Orbit Selection Summary

<table>
<tr><th rowspan="2">Evaluation Criteria</th><th colspan="5">Candidate Orbit Altitudes (km)</th></tr>
<tr><th>665</th><th>715</th><th>740</th><th>775</th><th>788</th></tr>
<tr><td>Shuttle Compatibility</td><td>Good</td><td>Good</td><td>Good</td><td>Good</td><td>Fair</td></tr>
<tr><td>Mission Compatibility</td><td>Good</td><td>Fair</td><td>Fair</td><td>Good</td><td>Good</td></tr>
<tr><td>Ground Station Compatibility</td><td>Poor</td><td>Fair</td><td>Good</td><td>Good</td><td>Good</td></tr>
<tr><td>Launch System Impacts</td><td>Good</td><td>Good</td><td>Good</td><td>Fair</td><td>Fair</td></tr>
<tr><td>Spacecraft Impacts</td><td rowspan="3" colspan="5">Minor impacts over range of altitudes</td></tr>
<tr><td>Payload Instrument Impacts</td></tr>
<tr><td>Impacts on Later Missions</td></tr>
</table>

selected orbit

SECTION 3.0

INSTRUMENT CONSTRAINTS AND INTERFACES

3.1 STUDY APPROACH

The instrument studies for EOS were conducted in the following general sequence:

a. Establish the mission/system requirements and the candidate instruments for EOS-A and -B
b. Establish the key issues and design drivers with respect to instrument requirements
c. Perform a technical evaluation of the candidate instruments for EOS-A and -B
d. Recommend instrument design modifications and generate instrument specifications where appropriate
e. Develop an EOS Instrument Interface Handbook (analogous to the NIMBUS Experimenters Handbook) to document spacecraft interfaces and data requirements
f. Evaluate instruments for future missions, highlighting the major problem areas and recommend possible solutions.

Various designs, centered around three basic scanning techniques, were evaluated for the Thematic Mapper. These were:

Scanner Technique	Instrument Study Contractor
Object Plan	Hughes
Image Plane, Linear	TE
Image Plane, Conical	Honeywell

Adaptions of each of these mechanical scanning techniques were also considered as candidate HRPI designs. In addition, two configurations of solid state electronically scanned detector arrays were also evaluated for HRPI based on studies performed by Westinghouse.

To provide maximum utility to the NASA from these various instrument studies, a uniform and more detailed set of baseline mission and performance requirements for each type of instrument was developed using the NASA study specification, EOS-410-02, for overall performance requirements. The candidate instrument designs were extrapolated to this

baseline, and then critiqued both in terms of their adaptation into the overall system and on a comparative basis relative to the alternate instrument design approaches.

For the MSS, studies focused on the adaptations required to operate that instrument in a lower orbit, compatible with future shuttle operations.

3.2 KEY ISSUES

Several key issues were defined and addressed as part of the instrument studies. These issues are reflected in the conclusions and recommendations and are summarized briefly.

Spacecraft Interface

The instruments are being developed to fly in the pre-shuttle era where the advantages of less emphasis on weight and volume are unfortunately not yet available. The use of a Delta launch vehicle requires light weight, small volume, and relatively low power instruments if more than one instrument is to fly on the same spacecraft.

Design Flexibility

The fluid state of definition of early EOS missions dictates that the basic instrument designs be flexible enough to accommodate changes in mission requirements (such as swath width, number and range of spectral bands, and/or operating altitude). These changes must be accommodated without major cost impact.

Design Risk

Both the high development cost and lack of shuttle retrieve capability during the early flights of the instruments require that the standard aerospace philosophy of minimum risk be followed. Risk in any of the areas of design, development, manufacture, test, launch environment, or lifetime must be given serious consideration in determining the acceptability of a particular instrument design.

System Performance

The remote sensing data user community has progressed to include sophisticated technical disciplinaries who are concerned with the utility of the data available. The "standard" performance parameters of resolution, geometric and radiometric accuracies by which

data quality is judged must be defined quantitatively and specified such that the user may determine the utility of the data for the type of information extraction that he requires.

3.3 CONCLUSIONS/RECOMMENDATIONS

The major study conclusions and design recommendations for the EOS-A instruments follow.

3.3.1 THEMATIC MAPPER

Scan Technique Selection

Any of the candidate scan techniques can be accommodated through ground processing to produce essentially equivalent quality in the output product. The cost differences of implementing any instrument scan data approach are small compared to total program cost. However, the conical scanner data processing will cost three to four hundred thousand dollars more than the other scanning approaches in order to linearize its data. Thus, the instrument technical evaluations must be combined with the instrument contractor's costs to make the final instrument recommendations and selection.

Radiometric and Geometric Accuracy

Specified radiometric and geometric accuracy will be difficult but nevertheless possible to achieve. The instrument portion of the total system's error budget has been developed. Initial response from the instrument contractors indicate only minor problems in meeting the budget.

Band-to-Band Misregistration

Band-to-band misregistration in the serial instrument data stream should be constrained to a range of several hundred pixels maximum in the cross-track direction only. Band 6 should lead the scan, since it requires fewer pixels storage to register to the other bands.

Number of Detectors Per Band

The number of detectors per band must be set with an integer relationship between bands 1-5 and band 6 to aid data registration. Reduced resolution (compacted data) must be an integral divisor of the total number of detectors in a band.

Spectral Separation Technique

Multilayer interference filters and spatial separation is preferred to prism monochromators for spectral band determination. The interference filters permit much flexibility in design and provide better optical efficiency. Along scan band-to-band registration can be met using good design and careful alignments. Alignment stability is most important, since fixed offsets can be provided for in-ground processing.

Maximum/Minimum Radiances

More detailed analysis of available data is required prior to the selection of maximum and minimum radiances.

3.3.2 MSS

The MSS is presently designed to provide 80 meter IFOV and a 15 Mbps data rate from its 914 km mission altitude. The simplest change to the MSS to allow operation with essentially the same resolution at any altitude between 700 and 914 km is to increase the IFOV to maintain the 80 meter pixel, alter the mirror sweep rate, and keep the data rate constant. Tradeoff studies indicate that it is least expensive to change optic curvatures to increase the IFOV than to change the internal fiber optics. The MUX needs to be modified to change certain timing relationships, and the scan mirror requires mechanical modifications to accommodate the new scan rates. The radiation cooler which was designed for the ERTS 9:30 A.M. orbit can be used for the EOS 11:30 A.M. orbit without any modifications. Electrical incompatibilities exist between the MSS and the EOS spacecraft but can be corrected either through an external interface box or internal MSS electronics redesign.

SECTION 4.0

DESIGN/COST TRADEOFFS AND RECOMMENDATIONS

The Design/Cost Tradeoff Studies report presents the rationale and results of the significant tradeoffs made during the first three months of the study and establishes the total program costs for the EOS-A mission. The report is organized into three major cost/trade areas: System, Spacecraft, and Ground Systems, with a fourth section on overall program costs.

4.1 SYSTEM DESIGN/COST TRADEOFFS

A series of mission/system level questions have been addressed through design/cost analyses. The results of these analyses have been organized to correspond to the cost tradeoff matrix presented in the study RFP and expanded upon in the General Electric proposal. Many of these system level cost trades are summaries of cost data developed from numerous detailed subsystem and lower level trade studies. The key conclusions from these studies follow.

- Orbit Altitude: 775 km (418 nm). The altitude has been selected to be compatible with both the EOS-A and EOS-B missions, as well as potential follow-on missions in both the pre- and post-shuttle era.
- Launch Vehicle: Delta 2910 for EOS-A and A'; Delta 3910 for a combined A and A' mission. Shuttle should be used for retrieve only (not resupply) for early missions.
- Shuttle Compatibility: Not a major problem or significant cost impact for a modular spacecraft design.
- Thematic Mapper and HRPI Approaches: All instrument versions must be size, weight, and power optimized. All can be accommodated; however, the conical scan approach requires higher equipment cost for ground data processing.
- Data Operations: A major consideration in that it varies program costs by several million dollars depending on number of users, system throughput and number of output products.

- Spacecraft Autonomy: An on-board processor such as the AOP can support all spacecraft computational functions and simplify certain ground functions. Size and speed of the AOP is adequate for planned spacecraft/ground function allocation.
- Electronic Technology: All spacecraft hardware can be implemented with existing or state-of-the-art designs to minimize cost and weight. The AOP memory, the command/telemetry remotes, and ground image processing buffer storage are areas where advanced technology may provide cost, size, weight, or power advantages in the future.
- Orbit Time of Day: Near noon with a typical descending node time of 1130 hours.
- Management Approach: Management approaches to minimize costs are defined. Absolute cost savings are difficult to quantify in most areas, but intuitively they appear real.
- Test Philosophy: A low-cost test philosophy is defined which minimizes test models and long, complete, system-level tests.
- Reliability and Quality Assurance: Two redundancy approaches considered: (1) to assure that no single failure would impair full mission success, and (2) to survive any single failure for subsequent Shuttle retrieval/servicing. Approximately $350,000 of redundancy primarily in category (2) is recommended in the C&DH, ACS, Propulsion and Wideband Systems.
- Commonality Potential: Commonality of hardware across multiple missions is practical. Multiple buys reduce cost. Minimum buy for five missions is recommended.
- International Data Acquisition: WBVTR and TDRSS approaches are roughly equivalent in recurring costs. The TDRSS approach involves higher development risk and higher non-recurring cost.
- Follow-on Instrument Accommodation: Major design drivers come from EOS-A. Maximum power subsystem capacity from SAR missions; ACS design affected by SEOS (geosynchronous) and Solar Max (near-inertial) missions. All can be accommodated with no major redesign impacts.

- System Requirements Allocation: Resolution, radiometric, and geometric performance requirements are difficult but can be met based on apportionment of requirements to the instruments and all subsystems.
- Spacecraft versus Ground Functions: Geometric correction of data for low cost users should be done on-board the spacecraft; all radiometric corrections performed on the ground. Parameters should be included in the wideband data stream to facilitate ease of ground processing.
- Spacecraft versus Shuttle Function: Initial spacecraft, using the Delta launch vehicle, should be shuttle retrievable. Titan and Shuttle launched spacecraft should be shuttle serviceable.
- Cost versus Weight and Volume: Tradeoff break point is $2,000 per pound. Key tradeoffs were made in propulsion, C&DH, thermal control, interstage adapter and power areas.

4.2 SPACECRAFT DESIGN/COST TRADEOFFS

This section summarizes the salient spacecraft and spacecraft subsystem cost trade results. The baseline designs recommended (see Section 6) reflect these tradeoff considerations. Key conclusions are:

- Structure: The general purpose spacecraft configuration is rectangular with simple structure of aluminum construction. Module size is 48 x 40 x 16 inches. A conventional aft adapter is used for either Delta or Titan launches. Shuttle retrieve capability is provided using a transition frame with three-point shuttle attachment.
- Thermal Control: Passive with heaters maintaining $\pm 5^{\circ}$ temperature range about the nominal.
- Propulsion: An integral hydrazine system is used for reaction control, orbit adjust and orbit transfer functions, with tank size selected for the specific mission application. Simple blowdown operation is used.
- Wideband: Design the EOS-A system for EOS-B (TM and HRPI) data rates. QPSK modulation is used for the wideband link; PCM/FM modulation for low-cost user link.

- Power: A regulated 28 ± 0.3 volt bus using direct energy transfer. Three batteries and regulators are used for EOS-A; maximum capacity of five will support SAR mission.
- ACS: Stellar (star tracker) and inertial sensors used for precise reference; control logic and control laws implemented in software for mission flexibility.
- C&DH: A two-bus data system used for either ground or AOP communication with spacecraft subsystems. Selected redundancy included. A narrowband tape recorder is used on early missions.

4.3 GROUND SYSTEM DESIGN/COST TRADEOFFS

Design/cost tradeoffs concentrated in five major ground system areas:

- Receiving stations and NASCOM facilities
- Operations Control Center
- Data Services
- Image Processing
- Low Cost Ground Stations.

The key conclusions are as follows:

- Antennas at Receiving Sites: Modify existing antennas for X-Band rather than procure new ones.
- OCC: Develop a new OCC capable of multi-vehicle operation and incorporating a shared disc for OCC/CDPF communications. Software and hardware are evolutionary design from ERTS, not entirely new developments.
- Data Management Element: Use for central control of all scheduling, image processing, product delivery, and management reporting in CDPF.
- Image Processing Element: An all new digital system is required to handle high throughput. Special purpose hardware or micro-processors are the only cost effective design approaches.
- Baseline Resampling/Processing Techniques: Use $\frac{\sin x}{x}$ with options for bilinear or nearest neighbor; baseline map projection is the Space Oblique Mercator with options for others. Product generation is done off line. A composite Browse

File and Extractive Processing Subsystem is used to minimize cost.

- Low Cost Ground Station: Consists of two standard subsystems plus user unique subsystem. Standard subsystems are Data Acquisition (for receipt and recording of data) and Data Processing and Correction (for radiometric correction of data; geometric corrections are done on board the spacecraft). These standard subsystems cost approximately $190,000. The third subsystem is user unique and typically consists of product generation and extractive processing equipment.

4.4 PROGRAM COST SUMMARY

Total program costs were developed for a combined operational/R&D mission using two identical spacecraft, EOS-A and A'. The payload is the 5-band MSS as the operational instrument while the Thematic Mapper is the R&D instrument. The spacecraft would be launched at one-year intervals with a two-year operating life. Each would be in 17-day repeat orbits. Two spacecraft provide 9-day coverage.

The program cost is predicated upon the following assumptions:

- Purchases of common hardware are to be made in minimum lots of five in order to take advantage of the cost savings in multiple buys.
- Minimum redundancy has been employed in the design of the spacecraft subsystems.
- The spacecraft and modules do not include hardware for shuttle on-orbit serviceability, but a modular design which can include these features in the future has been assumed.
- Costs for global coverage using WBVTR's are included.
- The power module and solar array are sized to deliver 200 watts orbit average power to the payload (in addition to basic spacecraft demands).
- X-band is used for all wideband communications to the ground.
- The central data processing facility will handle up to 175 scenes/day per sensor of Thematic Mapper/HRPI data.
- Recurring cost of the Low Cost Ground Station is estimated for a single unit assuming it to be one of ten produced.

- Spacecraft are to be launched using the Delta 2910.
- Launch dates for EOS-A and EOS-A' are one year apart (early 1979 and 1980, respectively).
- 1974 costs are presented— there has been no attempt to postulate the effects of inflation over the EOS mission model time span. Costs are presented through G&A; they do not reflect a contractor's fee.

4.4.1 CONCLUSIONS

As a result of the cost trades and analyses conducted during the EOS System definition study, the following conclusions can be reached.

- A low-cost basic spacecraft can be produced for a recurring cost of about seven million dollars.
- The Ground Data Handling System for an EOS mission that includes a TM and HRPI costs about $20 million assuming processing of about 175 scenes/day.
- A Low Cost Ground Station to receive data at a rate of 15 Mbps can be produced for a recurring cost of under $200K.

4.4.2 SPACECRAFT COST SUMMARIES

Table 4-1 presents the spacecraft costs for EOS-A. The costs of the basic spacecraft are separated from the costs of mission peculiar items and both non-recurring and recurring columns are shown. The basic spacecraft cost is for an integrated, tested spacecraft less all mission peculiars.

4.4.3 GROUND DATA HANDLING SYSTEM COST SUMMARIES

Table 4-2 shows the non-recurring and recurring costs for the Ground Data Handling System required to support the EOS-B spacecraft and to process the instrument data for dissemination to the users. Costs are shown for the OCC, the Central Data Processing Facility with a separate line item for annual operations costs. Network modifications and the Low Cost Readout Station costs are also included.

Table 4-1. EOS Spacecraft Cost Summary (Dollars X1000)

	Basic S/C		Mission Peculiar		EOS-A P/L: 1MSS/1 TM
	NR	R	NR	R	Remarks
Attitude Control System Module	5300	1400	—	—	
Power Module + Solar Array	3200	1200	—	—	
Communications & Data Handling Mod.	5500	1350	300	—	
Structure	100	50	220	100	
Thermal Control	—	—	—	—	Included in each module or S/S
Electrical Distribution	300	100	—	—	Harness incl. in each mod. or S/S
Interstage Adapter	—	—	150	50	
Propulsion Module	400	270	630	180	No retrieval capability
Wideband Module	—	—	5700	2900	Includes WB Gimbal, HDMR
Thematic Mapper Module	—	—	(12000)	(6000)	GFE
MSS Module	—	—	(2000)	(5000)	GFE
DCS	—	—	20	200	
Mechanisms	—	—	—	—	
Systems Level					
Program Management	1700	600	2100	1000	
Systems Engineering	3000	140	4000	300	
Pre-S/C Integration Test (BIT)	250	—	400	—	
System Integration (P/L)	—	—	600	100	Payload only
S/C Integration & Assy.	800	200	1200	300	
S/C System Test	2000	700	—	800	Includes SITE
Systems Test Equipment	1800	600	1500	800	
Reliability	400	—	700	—	
Quality Assurance	900	210	1200	500	
Documentation	220	100	500	250	
Launch Operations	200	—	500	300	
Sec. Services & T&L	700	200	170	900	
TOTALS	27270	7120	21320 (14000)	8680 (11000)	64,390 GFE

Table 4-2. EOS Ground Data Handling System Cost Summary (Dollars X1000)

Subsystem Level	NR	R	Remarks
Operations Control Center	2000	3300	
OCC Operations	—	1000	1 Year Operations
Central Data Processing Facility	7500	7500	
CDPF Operations		2500	1 Year Operations
Network Modifications	1530	—	3 Sites
Low Cost Readout Station	650	190	Assumes On-board Spacecraft Correction

The costs include all hardware required, program management, system engineering, spares, system integration and test, reliability, quality assurance, documentation, operations support, support services, user services, secretarial support and travel and living.

4.4.4 MISSION COST SUMMARY

The EOS-A mission cost summary is shown in Table 4-3. The launch vehicle used in this cost summary is the Delta 2910 and the spacecraft has been designed weight-wise with this capability in mind.

Table 4-3. EOS Mission Cost Summaries (Dollars X1000)

Item	EOS-A NR	EOS-A R
Basic Spacecraft	27270	7120
Mission Peculiars	21320	8680
Spacecraft Totals	48590	15800
Operations Control Center	2000	3300
OCC Operations	—	1000
Central Data Processing Facility	7500	7500
CDP Operations	—	2500
Network Modifications	1530	—
Ground Systems Totals	11030	14300
Launch Vehicle		6000
Sub-totals	59620	36100
Total Mission Cost (Less Instruments)	118670	

SECTION 5.0

LOW-COST MANAGEMENT APPROACH AND RECOMMENDATIONS

The EOS Program as configured by NASA provided an opportunity to look at the elements of cost commonly labeled "Management," to determine if there is a better, more economical way of doing business to further reduce total program costs. The cost reduction trend has been evident in the aerospace industry for several years but has been primarily directed toward the traditional cost improvement approach. In order to develop a true low-cost approach, the factors which cause cost to be incurred must be identified — the so-called "cost drivers." The process of identifying and evaluating cost drivers soon indicated that the best approach to "low cost" is good, sound business management practices by both NASA and Industry.

This approach has lowered the costs of commercial products and will reduce the cost of aerospace products. To lower costs means to cut out the "fat," minimize inefficiencies, and simplify. In summary, a low EOS Program cost demands matching all requirements to expected performance; identifying reasonable risk for both NASA and Industry, establishing a cost, and managing to that "cost" by both parties.

In this study, several assumptions were made:

- "Business as usual" can be sufficiently defined to serve as a bench mark for showing lower cost in the techniques resulting from this study.
- Reductions in NASA's "business as usual" requirements of any kind can be made if a lower cost can be shown and justified as not impairing the required performance or increasing the total program risk beyond acceptable levels.
- A defined risk can be mutually agreed upon and jointly borne by both NASA and Industry, where experience says that the cost savings justified the increased risk.

The key conclusions of this study were the following:

- Systems and procedural controls imposed on the fledgling aerospace industry to date should be relaxed on the now more mature aerospace industry.

- Commonality can significantly reduce hardware costs by allowing multiple instead of single buys.
- CDRL's should be minimized to necessary and sufficient documentation by eliminating "nice-to-have" requirements.
- A two-phase contract is recommended as the most cost-effective contracting technique; a CPIF/AF development phase, with a follow-on fixed price production phase with successive incentive targets.
- Traditional concepts of extensive testing at all levels of assembly can be relaxed particularly in the era of on-orbit repair and retrieval since must malfunctions and failures lose their catastrophic implications.
- Reliability and quality programmatic requirements can and should be simplified.
- Design-to-cost techniques have applicability to low-cost space programs.
- Value Management is a cost incentive which should be used, typically, during the design phase.

5.1 OBJECTIVE

It was not an objective of this study to analyze and describe how industry could perform internally at a lower cost, but to identify and define Industry/NASA interfaces that would produce a lower EOS Program cost than if the program were implemented on a "business as usual" basis. Therefore, the study was directed at the NASA/Industry interface and how that interface could be improved so that NASA and Industry's internal implementation can, as a result, be simplified and more cost effective.

5.2 APPROACH

To establish the basic framework of this study task, an experienced team of senior GE management personnel was assembled to identify the management areas or techniques which, in their judgment, offered a good potential for cost reduction. The areas identified included all of those recommended by NASA in the RFP and the following additional innovative concepts:

- EOS commonality potentials.
- The possible application of a "Design-to-Cost" philosophy and phased contracts.

- The possible application of appropriate commercial practices to aerospace contracts.
- The possible contractual application of Value Management.

5.3 RECOMMENDATIONS

5.3.1 PROGRAM MANAGEMENT

This portion of the study effort was concerned solely with the interaction between NASA and industry in the areas of generalized program management and control. The question was asked, "How best and most economically can NASA maintain the required control of industry while maintaining the orderly progression of effort?" Although it is not possible to quantize the cost savings, intuitively it is apparent that some savings in EOS Program Management costs would result if industry is provided less regimentation and adherance to rigid check and balance systems of control which prescribe how the contractor shall perform but add little value to the products. The maturity of the aerospace product has reached a position in EOS where more reliance can now be placed on the capability of the aerospace industry and, therefore, the tight contractor control that has developed over the past years can, in fact, be somewhat relaxed.

5.3.2 COMMONALITY POTENTIAL

The potential for cost savings of common hardware in the EOS program was summarized in this study task based upon the results of the Design/Cost Tradeoff Studies described in Report No. 3. This summary resulted in a listing of common hardware items for which required quantities were developed using the EOS Study Mission Model, reviewed for shelf life impact, and multiple buy savings. From this summarization the following recommendations were developed:

- Design the General Purpose spacecraft to use the same hardware to perform multiple missions.
- Make multiple buys of hardware with a minimum purchase of sets for at least five spacecraft.
- Provide controlled storage environments, and exercise and retrofit selected components as required. (Shelf life of five years for spacecraft hardware does

not appear to be a problem based upon previous studies conducted.)

5.3.3 CONTRACTING TECHNIQUES

This area was specifically recommended by NASA for analysis as part of the study contract. However, a review of available data indicated no clear cut case of lower cost for any particular contracting technique. Therefore, the resulting recommendations from this study analysis focused on the most cost effective technique at the NASA-Industry interface. It was recommended that the most cost-effective technique would be a Prime Spacecraft Systems Contract with Associate Instrument contracts. This contracting technique would accomplish the following:

- enable GSFC to trade cost, schedule, and performance between the instruments and the spacecraft system,
- utilize NASA's on-board expertise to fullest advantage in instrument development,
- reduce the NASA-S/C contractor — instrument contractor interfaces,
- provide effective contractual control over the minimum required number of contractors,
- allow initiation of firm instrument development contracts before initiation of the prime system contract as required by development cycles.

The prime spacecraft system contract would be most cost effective as a two-phase contract; a cost plus development phase, and a fixed price production phase. This two-phased approach would allow the use of two innovative techniques that have proven very successful in cutting costs in GE commercial products; namely, design-to-cost and value management, both of which require and/or work best in a phased environment.

5.3.4 TEST PHILOSOPHY

An approach was developed by taking a hard look at present day testing techniques, applying only those needed to an early EOS system and then modifying that approach based upon multiple missions utilizing identical spacecraft bus hardware, fully modular design, on-orbit repair and on-board computer utilization for test and troubleshooting. This produced an EOS test philosophy which reduced spacecraft level tests to a minimum placing greater

emphasis on comprehensive environmental testing at the subsystem level, while system testing is relegated to the role of "workmanship" and go/no-go tests.

Since the EOS program will be a multiple vehicle program utilizing the same basic subsystem modules and structure for each spacecraft, it is uniquely suited for such an approach. The subsystem modular concept also lends itself to this philosophy. Subsystem environmental testing at the module level can be made as fully stringent and realistic as at the spacecraft level. Further, any subsequent module replacement due to malfunction or failure during systems testing can be made with minimum impact on the spacecraft test program because environmental testing has already taken place.

5.3.5 RELIABILITY

Reliability requirements for NASA programs are generally specified by NHB 5300.4, "Reliability Program Provisions for Aeronautical and Space System Contractors." April 1970, and are applied in totality or by specific paragraphs only. During the many years of implementation of the provisions of this document, it has been found that certain of these tasks make a significant contribution to the removal of unreliability from a space system, whereas other tasks have little or no impact on the hardware at all and can be eliminated with little risk to the program and with a consequent cost savings.

The recommended reliability program would respond only to the provisions of NHB 5300.4 which are considered necessary to eliminate or alleviate the major and sometimes subtle failure modes from the satellite system and deletes those having little program value. The program would consider not only the selected contractors task responsibilities, but also recommend the inclusion of certain provisions in the NASA Statement of Work (SOW) that can influence the NASA/Contractor interface and task responsibilities.

5.3.6 QUALITY ASSURANCE

The same approach as in 5.3.5 above was taken on analyzing Quality Assurance since both Reliability and Quality are covered by the same document (NHB 5300.4). Each of the defined tasks in NHB 5300.4 (1B) is a required element in any space-oriented program;

however, certain modifications to these provisions would result in a more cost-effective Quality Program.

In summary, the elimination of any tasks in their entirety is not recommended, but modification of the following tasks is recommended:

1B103	Quality Program Documents
1B204	Quality Status Reporting
1B300	Technical Documents
1B302	Change Control
1B502	Procurement Documents
1B504	Government Source Inspection
1B801	Nonconformance Documentation
1B804	Material Review Board
1B806	Supplier Material Review Board.

The modifications are discussed and explained in detail in Report No. 4 and implementation of these modifications by NASA would produce a lower total program cost than if Quality Assurance and Reliability were conducted in a "business-as-usual" manner.

5.3.7 PROGRAM DOCUMENTATION

A reduction in a "business-as-usual" Contract Data Requirements List (CDRL) of 75% was accomplished by application of the following criteria:

- Availability of information at the contractor's facility for customer persual rather than required submission.
- Maximum combination of reports to reduce redundant efforts.
- Use of contractor internal documentation whenever possible.
- Use of multi-detail drawings.
- Use of red-line and/or preliminary drawings in development phase.
- Use of existing NASA-approved documents applicable to EOS.
- Reduce depth and frequency of financial and progress reports.
- Maximize exception reporting to minimize cyclic reports.
- Reduce number of copies submitted to essentials.

What remained is a list of necessary and sufficient documentation for NASA Management of an EOS program. This was estimated at almost a 1.0% total program cost savings.

5.3.8 DESIGN-TO-COST

The greatest single difference between a spacecraft and commercial product development and fabrication program is relative unit cost. A method of keeping the commercial product cost low is "Design-to-Cost" which is an iterative process whereby hardware and services are provided within the total cost constraints established by customer or market. In the simplest sense, the price which the market or consumer is willing to pay for the product dictates what the selling price will be. Performance, reliability, and quality are traded down to meet the cost goal.

To apply design-to-cost in a spacecraft program like EOS appears feasible provided that NASA plans a two-step procurement; a design-to-cost phase and an implementation phase. During the design phase:

- Required performance envelopes are defined;
- Performance characteristics and levels which influence fabrication, test, launch, and operations costs are determined;
- Design configurations are costed on a life cycle basis; and
- Realistic cost goals are established as a function of performance.

Then, as a result of the Design-to-Cost phase, the design implementation phase of the contract is consummated at the cost goals established by NASA and Industry. This two-step procurement is not unlike the practices of a competitive design contract followed by a production contract. What is new is that NASA and Industry would jointly select modification of those performance parameters which can reasonably be accepted prior to and during the shuttle era so as to minimize the spacecraft life cycle cost.

5.3.9 VALUE MANAGEMENT

Other cost trade studies of Value Management have indicated a significant reduction in proposed program costs by initiating this technique as early as possible in a program, (even as early as the RFP response). Therefore, based upon the contracting technique recommended earlier (namely, phased contracts) initiation of Value Management in Phase I should result in lower cost EOS designs.

For the EOS Program, it is recommended that NASA incorporate a "Program Clause" for Value Management, providing for $100,000 of specific effort. It is also recommended that NASA contribute to the selection of the two to four studies to which this funding should be allocated.

SECTION 6.0

BASELINE SYSTEM DESCRIPTION AND SPECIFICATIONS

6.1 INTRODUCTION

This section summarizes the recommended system baseline design established to satisfy the requirements of the next generation of Earth Observatory Satellite missions. The first mission (EOS-A) is envisioned as a two-fold mission which (1) provides a continuum of data of the type being supplied by ERTS for the emerging operational applications and also (2) expands the research and development activities for future instrumentation and analysis techniques. The baseline system specifically satisfies the requirements of this first mission. However, EOS-A is expected to be the first of a series of earth observation missions. Thus the baseline design has been developed so as to accommodate these latter missions effectively as the transition is made from conventional, expendable launch vehicles and spacecraft to the era of the Shuttle Space Transportation System. Further, alternative mission requirements including Seasat, SEOS, SMM, and MSS-5 have been analyzed to verify that the basic spacecraft design can also serve these missions, thus demonstrating that a multi-mission spacecraft design is economically sound.

A key feature of the baseline system design is the concept of a modular observatory system whose elements are compatible with varying levels of launch vehicle capability. The design configuration can be used with either the Delta or Titan launch vehicles and will adapt readily to the Space Shuttle when that system becomes available in the early 1980's. The ability to match various launch vehicles to the required spacecraft weight and altitude for a given mission using a common multi-purpose spacecraft greatly improves mission economy and flexibility.

Commonality of the basic spacecraft modules for multiple mission use has been adopted to achieve low total costs. This concept utilizes a set of basic service subsystems whose design and performance support a variety of missions without redesign. By standardizing the mechanical configurations and electrical interfaces of the subsystem modules, and by designing each of them to be structurally and thermally independent entities, they have

been configured to support mission-unique instruments and other payloads without redesign.

The modularity concept has been extended to provide for eventual on-orbit replacement of elements using the Space Shuttle in the 1980's. On-orbit service can be used for periodic maintenance or for replacement in case of failures. In addition, the spacecraft is retrievable by the shuttle for refurbishment on the ground. This further extends the economic benefits of the system design in the shuttle era.

6.2 MISSION REQUIREMENTS AND SYSTEM DESCRIPTION

The baseline system design was evolved after a series of design/cost tradeoffs against a set of mission/system requirements and guidelines provided by GSFC. This section summarizes the principal or driving requirements and the salient system characteristics which have evolved.

6.2.1 EOS-A MISSION REQUIREMENTS

The initial mission in the EOS series is a combined operational/R&D mission. The operational mission uses the developed 5-band MSS as the principle instrument to provide data continuity following ERTS-C to support the ongoing emerging operational applications. The R&D mission is oriented to Land Resource Management development. This mission will develop advanced instruments and processing systems which can provide multispectral imagery of the land surface of the earth at significantly improved spatial and spectral resolutions than the operational data. The object plane scanning Thematic Mapper is used as the baseline R&D instrument.

The principle EOS-A requirements and guidelines are outlined in Tables 6-1 and 6-2. Table 6-1 indicates those requirements principally resulting from cost trades. Table 6-2 is the requirements or guidelines provided by GSFC during the study. Although many are not quantitative, they reflect the principle requirements against which the EOS-A baseline system has been developed.

Table 6-1. EOS-A System Requirements (from Cost Trades)

PARAMETER	SYSTEM REQUIREMENT
Orbit	• Compatible With Both EOS-A and EOS-B Missions • Direct Shuttle Access • Sun-synchronous • Node Time: 1100 - 1130
Spacecraft	• Use General Purpose S/C • Modularity at Subsystem Level • Conventional Launch Vehicle
Shuttle Utilization	• Retrieval Only (at 612 km altitude) • Back-up Retrieve at Mission Altitude
Data to Local Users	• Provide Low Cost Receive/Process Capability

Table 6-2. System Guidelines

CHARACTERISTICS	GUIDELINE
Launch Vehicle	Delta 2910
7-9 Day Repeat Cycle	Achieve With Two Spacecraft
Global Data Return	TDRSS With WBVTR Alternate
Thruput	10^{10} - 10^{12} Bits/Day
STDN Ground Stations	Alaska, NTTF, Goldstone
Timeliness of Data Delivery	Consistent With DOMSAT Relays Between: • Acquisition Stations and GSFC • GSFC to Sioux Falls
Mission Duration	2 Years
Swath Width	185 km
Control Center	At GSFC
Data Processing	• Design for EOS-B • Operate Initially TM Only
Frequency Allocations	Wideband • TDRSS - Ku • Direct - X C&DH - S

6.2.2 OTHER MISSION REQUIREMENTS

The Earth Observatory Satellite (EOS) Program concept includes an economical, multi-purpose, modular spacecraft design which is compatible with many mission requirements. Typical missions that could utilize the EOS spacecraft are SMM, Seasat, ERS, and SEOS.

The matrix in Table 6-3 summarizes the various missions considered and identifies the salient requirements that each imposes on the multi-purpose spacecraft design. The relative influence of the mission matrix on the spacecraft design and performance is most sensitive for the EOS series of missions. This results largely from the type of instruments required for these missions. The commonality of requirements across each row of the matrix are evident and have been used to set subsystem functional and performance requirements to satisfy the entire mission matrix cost effectively.

6.2.3 EOS SYSTEM CONFIGURATION

The overall system configuration applicable to the EOS-A and follow-on missions is illustrated in Figure 6-1. The observatory uses the standard modular spacecraft bus to accommodate the MSS and TM instruments and related mission peculiar equipment. Two pointable X-band antennae transmit TM and MSS data direct to STDN or International ground stations. The two identical links facilitate station handovers and provide the capability to transmit full data to more than one station simultaneously as might be required, for example, over North America when transmitting to both the Alaska (STDN) and Prince Albert (Canadian) Stations. Combined TM and MSS data may also be transmitted at Ku-band through an unfurlable 8-foot oriented dish and relayed via TDRS when the spacecraft is beyond the view of one of the STDN stations. After recording at the ground station, payload data may be played back at reduced rate and relayed to GSFC for processing via DOMSAT. Physical delivery of tapes is available as a backup. In addition, selected reduced bandwidth TM data (compacted data) is transmitted via a fixed earth oriented shaped beam antenna to any of the low-cost local user stations.

Spacecraft telemetry, tracking, and command data are transmitted and received at S-band. Full two-way capability exists either through the spacecraft omni antenna and the STDN

Table 6-3. EOS Multiple Mission Data

<table>
<tr><th colspan="3" rowspan="2">SYSTEM</th><th rowspan="2">EOS A & A'</th><th colspan="4">EOS FOLLOW-ON</th><th rowspan="2">SHUTTLE RESUPPLY DEMONSTRATION TEST FLIGHT</th></tr>
<tr><th colspan="2">OCEANOGRAPHY METEROLOGY MISSION</th><th colspan="2">ALL WEATHER OBSERVATORY</th></tr>
<tr><td colspan="3">MISSION</td><td>PROVIDE CONTINUED OPERATION GATHERING OF EARTH RESOURCES DATA USING THE MSS INSTRUMENT. DEVELOP ON ADVANCED INSTRUMENT WHICH CAN PROVIDE MULTIPSECTRAL IMAGERY OF THE LAND SURFACE AT SIGNIFICANTLY IMPROVED SPATIAL AND SPECTRAL RESOLUTIONS OVER ERTS. STUDY DIRECTION IN WHICH OPERATIONAL LAND USE INVENTORY AND EARTH RESOURCE MANAGEMENT PROGRAMS SHOULD PROCEED.</td><td colspan="2">PERFORM RESEARCH IN THE PRIORITY PROBLEMS OF OCEANOGRAPHY AND METEOROLOGY, ESPECIALLY THOSE ASSOCAITED WITH AN IMPROVED DATA BASE FOR LONG RANGE WEATHER FORECASTING AND FOR OCEAN RESOURCE MODELING.</td><td colspan="2">DEVELOP ALL WEATHER CAPABILITY FOR BOTH ATMOSPHERIC STRUCTURE DETERMINATION AND SURFACE OBSERVATION.</td><td>VERIFY EOS COMPATIBILITY WITH THE SHUTTLE CAPABILITY FOR LAUNCH RESUPPLY AND RETRIEVAL. FINAL "SHAKE-DOWN" OF COMBINED SHUTTLE, THE RESUPPLIABLE OBSERVATORY AND THE FLIGHT SUPPORT SYSTEM.</td></tr>
<tr><td colspan="3">PAYLOAD</td><td>MULTISPECTRAL SCANNER
(SCANNER: 1.77 x 1.95 x 1.42; 127)
(MULTIPLEXER: 0.33 x 0.5 x 0.54; 7.5)
THEMATIC MAPPER
(x x ; 320)
DATA COLLECTION SYSTEM
(1 x 1 x 2; 77)</td><td colspan="2">PASSIVE MULTICHANNEL MICROWAVE RADIOMETER
(*79 FT^3; 200)
ADVANCED ATMOSPHERIC SOUNDER
(1.5 x 2.3; 100)
RADIOMETRIC SCATTEROMETER
(ELECTRONICS: 1 x 1 x 1
ANTENNA: 3.28 dia x 1.64; Total wts = 115)
LIMB ATMOSPHERIC COMPOSITION PROFILIER
(1.33 dia x 5 1/4; 168)
ADVANCED MAPS
(1.2 x 0.75 x 0.67; 80)
OCEAN SCANNING SPECTROPHOTOMETER
(2.16 x 1.46 x 0.75; 60)</td><td colspan="2">SYNTHETIC APERTURE RADAR
(ANTENNA: 27 x 2.5 x 1;
ELECTRONICS: 5.1 FT^3;
TOTAL WEIGHT = 387 LBS)
THEMATIC MAPPER
(3 x 3 x 7, ≤ 600)</td><td>* ENGINEERING MODEL HARDWARE OR BACKUP PAYLOADS FOR EOS-A.</td></tr>
<tr><td rowspan="3">ORBIT</td><td colspan="2">ALTITUDE</td><td>418* nm</td><td colspan="2">450 nm</td><td colspan="2">418 nm</td><td>300 nm</td></tr>
<tr><td colspan="2">INCLINATION</td><td>98.5° Sun Synchronous</td><td colspan="2">98.76° SUN SYNCHRONOUS</td><td colspan="2">98.5° SUN SYNCHRONOUS</td><td>28.5°</td></tr>
<tr><td colspan="2">ACS MODE TIME</td><td>2330</td><td colspan="2">1200</td><td colspan="2">2330</td><td>NOT CRITICAL</td></tr>
<tr><td rowspan="2">POWER</td><td colspan="2">TYPE</td><td>ONE SINGLE AXIS ORIENTED SOLAR ARRAY</td><td colspan="2">ONE SINGLE AXIS ORIENTED SOLAR ARRAY</td><td colspan="2">ONE SINGLE AXIS ORIENTED SOLAR ARRAY</td><td>ONE SINGLE AXIS ORIENTED SOLAR ARRAY OR BATTERY POWER</td></tr>
<tr><td colspan="2">POWER LEVEL</td><td>500 w AVERAGE</td><td colspan="2">550 w AVERAGE</td><td colspan="2">** 450 w AVERAGE</td><td>500 w AVERAGE</td></tr>
<tr><td rowspan="5">ACS</td><td colspan="2">REFERENCE</td><td>STELLAR</td><td colspan="2">STELLAR</td><td colspan="2">STELLAR</td><td>STELLAR</td></tr>
<tr><td colspan="2">TYPE</td><td>3 AXIS, ZERO MOMENTUM</td><td colspan="2">3 AXIS, ZERO MOMENTUM</td><td colspan="2">3 AXIS, ZERO MOMENTUM</td><td>3 AXIS, ZERO MOMENTUM</td></tr>
<tr><td rowspan="3">ACCUR.</td><td>POINT</td><td>0.007 DEG</td><td colspan="2">0.05 DEG</td><td colspan="2">0.05 DEG</td><td>0.007 DEG</td></tr>
<tr><td>RATE</td><td>5×10^{-5} DEG/SEC</td><td colspan="2">6.7×10^{-4} DEG/SEC</td><td colspan="2">3.6×10^{-4} DEG/SEC</td><td>5×10^{-5} DEG/SEC</td></tr>
<tr><td>KNOW.</td><td>0.007 DEG</td><td colspan="2">0.05 DEG</td><td colspan="2">0.05 DEG</td><td>0.007 DEG</td></tr>
<tr><td rowspan="2">WIDE-BAND DATA</td><td colspan="2">RATE</td><td>135 Mbps MAX</td><td colspan="2">2.5 MBPS</td><td colspan="2">TWO 120 MBPS CHANNELS</td><td>NO</td></tr>
<tr><td colspan="2">ON-BOARD STORAGE</td><td>OPTION TO TDRS</td><td colspan="2">OPTIONS TO TDRS</td><td colspan="2">OPTIONS TO TDRS</td><td>TELEMETRY ONLY</td></tr>
<tr><td colspan="3">LAUNCH VEHICLE</td><td>DELTA 2910</td><td colspan="2"></td><td>* DELTA</td><td>TITAN</td><td>SHUTTLE-DIRECT</td></tr>
<tr><td rowspan="3">TUG</td><td colspan="2">TYPE</td><td>INTEGRAL</td><td colspan="2">INTEGRAL</td><td>INTEGRAL</td><td>INTEGRAL</td><td>NOT REQUIRED</td></tr>
<tr><td colspan="2">PROP. TYPE</td><td>HYDRAZINE</td><td colspan="2">HYDRAZINE</td><td>HYDRAZINE</td><td>HYDRAZ/SOLIDS</td><td>--</td></tr>
<tr><td colspan="2">NEED FOR ORBIT ADJU.</td><td>YES</td><td colspan="2">NO</td><td>YES</td><td>YES</td><td>NO</td></tr>
<tr><td rowspan="3">SPACECRAFT CHARACTERISTICS</td><td colspan="2">WEIGHT ON ORBIT</td><td>2350</td><td colspan="2">2400 LBS</td><td>2500 LBS</td><td>4000 LBS</td><td>6500 LBS</td></tr>
<tr><td colspan="2">LENGTH</td><td>16 FT</td><td colspan="2">18 FT</td><td>18 FT</td><td>21 FT</td><td>27 FT</td></tr>
<tr><td colspan="2">DIAMETER</td><td>7 FT</td><td colspan="2">7 FT</td><td>7 FT</td><td>9 FT</td><td>9 FT</td></tr>
<tr><td colspan="3">LAUNCH DATES</td><td>EOS-A - 1979
EOS-B - 1980</td><td colspan="2">1980</td><td colspan="2">1981</td><td>1980</td></tr>
<tr><td rowspan="2">LIFETIME</td><td colspan="2">LIFETIME</td><td>2 YEARS</td><td colspan="2">2 YEARS</td><td colspan="2">2 YEARS</td><td>7 DAYS</td></tr>
<tr><td colspan="2">RETRIEVE</td><td>RETRIEVE 1983</td><td>RETRIEVE 1983</td><td>RETRIEVE/REFURBISH 1983</td><td>RETRIEVE 1983</td><td>RETRIEVE/RESUPPLY 1983</td><td>RETRIEVE IS PART OF MISSION</td></tr>
<tr><td colspan="3">NOTES</td><td>* EOS-A PROVIDES 17 DAY REPEAT CYCLE
EOS-A' TO BE PHASED WITH EOS-A TO PROVIDE 8/9 DAY REPEAT</td><td colspan="2">*INCLUDES TWO 3.6 FT DIAMETER SCANNING ANTENNAS.
**DELTA CONFIGURATION WILL CARRY THE PMMR, AS MANY OTHER INSTRUMENTS AS POSSIBLE BUT NOT THE RADIOMETER SCATTEROMETER.</td><td colspan="2">* DELTA 2910 CONFIGURATION WILL CARRY ONLY THE SAR; DELTA 3910 REQUIRED FOR SAR PLUS TM.
**1250 WATTS PEAK POWER FOR 10 MIN REQUIRED FOR THE SAR.</td><td>*ASSUME PAYLOADS TO BE TURNED ON AND IMAGERY ACQUIRED TO SUPPORT FULL UP SHUTTLE AND SPACECRAFT COMPATIBILITY TEST MISSION. MORE LIKELY MISSION WILL HAVE MINIMUM S/C CAPABILITY (NO OPERATING PAYLOADS, NO SOLAR ARRAY)</td></tr>
<tr><td colspan="3">REFERENCES</td><td>EOS SYSTEM DEFINITION STUDY RESULTS</td><td colspan="2">INTERNAL GE CONCEPTUAL EOS-B PAYLOAD</td><td colspan="2">INTERNAL GE CONCEPTUAL EOS-C PAYLOAD</td><td>"EOS REQUIREMENTS FOR EARLY SHUTTLE FLIGHTS", GSFC MAY 1973</td></tr>
</table>

SEOS	SOLAR MAX	SEASAT-A	SEASAT-B
DEVELOP REMOTE SENSING TECHNOLOGY FOR MEASUREMENT OF EARTH'S TRANSIENT ENVIRONMENT FROM SYNCHRONOUS ALTITUDE	INVESTIGATE FLARES AND RELATED PHENOMENA AND THEIR EFFECTS ON THE SOLAR-TERRESTRIAL SYSTEM THROUGH A WELL COORDINATED SET OF UNIQUE INSTRUMENTATION FOR OBSERVING TRANSIENT ULTRAVIOLET, HIGH-ENERGY AND VISIBLE RADIATION.	GLOBAL SCALE MONITORING OF WIDE RANGE OF PHYSICAL OCEAN PHENOMENA; SEA STATE CURRENTS, CIRCULATION, TIDES, WIND STRESS AND GEOID UNDULATIONS. DEMONSTRATE KEY FEATURES OF OPERATIONAL SYSTEM.	GLOBAL SCALE MONITORING OF WIDE RANGE OF PHYSICAL OCEAN PHENOMENA; SEA STATE CURRENTS, CIRCULATION, TIDES, WIND STRESS AND GEOID UNDULATIONS.
4.9 FT DIAMETER (CASSAGRAIN) TELESCOPE ASSY. WITH VISIBLE, NEAR IR & THERMAL IR DETECTORS. RESOLUTION BETTER THAN 100 m VISIBLE AND NEAR IR, APPROX. 1000 m IN THERMAL IR. (TELESCOPE: 6.55 dia. x 13.1; 1144) (SENSOR ASSY: 6.73 dia x 3.3; 320) DATA COLLECTION SYSTEM (ANTENNA: 1.24 x 1.24 x 1; 31) (ELECTRONICS VOL: 1.32 FT^3; 44)	MINIMUM PAYLOAD: UV MAGNETOGRAPH (0.58x0.84x6; 100) EUV SPECTROMETER (0.84x0.84x6; 100) HIGH RESOLUTION X-RAY SPECTROMETER (0.54 x 0.84 x 6.5; 100) HARD X-RAY IMAGING (0.5x0.42x6.5,100) LOW/MEDIUM X-RAY POLARIMETER (0.67 x 0.67 x 3; 16) GAMMA RAY DETECTOR (1.5x1.5x3; 200) H-ALPHA PHOTOMETER (0.33x0.33x3; 20) FLARE FINDER (0.33 x 0.33 x 6; 30) ADDITIONAL SENSORS: IN ORDER OF IMPORTANCE ARE: HARD X-RAY SPECTRO. (1 x 1 x 3; 70) SOLID STATE X-RAY DETEC. (1x1x1; 20) CORONOGRAPH (0.42 x 1 x 6; 100) UV SPECTROMETER (0.67 x 1 x 6; 110) NEUTRON DETECTOR (0.83x1.67x3; 205)	RADAR ALTIMETER (3.28 DIA; 99)* 5-CHANNEL MICROWAVE SCANNING RADIOMET. (4.1 DIA; 110)* DUAL FREQUENCY SCATTEROMETER (5 at 8.8 LENGTH; 385)* VISIBLE & SCANNING RADIOMETER (2 x 3 x 2; 22)* ** SAR	ALTIMETER - K-BAND (0.66 x 0.66 x 1.64; 100) SCATTEROMETER - K-BAND (3.6 x 4.92 x 3.28; 200) IR SCANNER (3.28 x 3.28 x 3.96; 95) SATELLITE TO GROUND TRANSPONDER (0.82 x 0.66 x 0.66; 17.6) SATEL.-TO-SATEL. TRANSPONDER (1 x 2 x 2; 88) RETRO REFLECTORS (2 x 0.1 x 1.5; 44) COHERENT RADAR ALTIMETER (1 x 3.28 x 3.28; 161) *SAR
19,323 nm	285 nm	430 nm	324 nm
2° GEOSYNCHRONOUS	30°	108 DEG	90°
N/A - POSITIONED AT 96° W. LONGITUDE	N/A	N/A	N/A
SINGLE AXIS ORIENTED SOLAR ARRAY	DUAL FIXED ARRAY	ORIENTED SOLAR ARRAY (TWO AXIS)	FIXED BODY MOUNTED SOLAR CELLS, OR SINGLE AXIS ARRAY + FIXED ARRAY, OR 2 AXIS ORIENTED ARRAY
400 w AVERAGE	235 w AVERAGE	465 w	375 w AVERAGE
STELLAR	SOLAR/STELLAR	EARTH	EARTH
3 AXIS, ZERO MOMENTUM	* 3 AXIS, ZERO MOMENTUM	3 AXIS, ZERO MOMENTUM	3 AXIS GRA GRAD. WITH MOMENTUM WHEEL SUN & HORIZON SENSORS
0.017 DEG	1.2 SEC WITH FLARE FINDER; 1 MIN W/OUT	0.5 DEG	2 DEG
≈ 10^{-5} DEG/SEC	1 SEC OVER 5 MIN WITH FLARE FINDER	--	0.002 DEG/SEC
0.0017 DEG	< 5 SEC FROM FLARE FINDER	0.2 DEG	0.1 DEG
10 MBPS	128 KBPS	**1.6 MBPS	90 MBPS
NO	YES (OR TDRS)	YES	YES
SHUTTLE	DELTA	DELTA	SHUTTLE-DIRECT
NON INTEGRAL SPACE TUG	NOT REQUIRED	NOT REQUIRED	NOT REQUIRED
HYDRAZINE	—	—	—
YES	NO	YES	NO
2716 LBS	*** 2534 LBS	2050	2230 LBS
22.7 FT	** 14 FT	13 FT	15 FT
10.7 FT	** 7 FT	7 FT	13 FT
1981	1978	1977	1982
2 YEARS	2 YEAR	1 YEAR	5 YEARS
RESUPPLY EVERY TWO YEARS	RETRIEVE EVERY TWO YEARS	NO RETRIEVE	NO RETRIEVE
SYSTEM MUST BE CAPABLE OF PROVIDING TEN 70 MILIRADION (4°) SCANS PER HOUR TO SCAN THE EARTH AT CONSTANT SUN ELEVATION.	*10 MIN SLEW IN ONE AXIS FOLLOWED BY 10 MIN SLEW IN SECOND AXIS IN 30 SECONDS TOTAL IS REQUIRED. SLEW BASED ON ERROR SIGNALS FROM FLARE FINDER SENSOR. **END VIEWING CONFIGURATION. ***WEIGHT CORRESPONDS TO MINIMUM PAYLOAD.	* SIZES OF ANTENNAS ONLY, WEIGHTS ARE ENTIRE SUBSYSTEMS ** POTENTIAL ADDITION OF SAR COULD HAVE MAJOR IMPACT ON SENSOR COMPLIMENT AND SPACECRAFT CONFIGURATION. *** DOES NOT INCLUDE PROVISIONS FOR SAR	*POTENTIAL ADDITION OF SAR COULD HAVE MAJOR IMPACT ON SENSOR COMPLIMENT AND SPACECRAFT CONFIGURATION.
"SSPD DATA SHEETS, EO-09-A" 10/9/73	"SOLAR MAXIMUM MISSION CONCEPTUAL STUDY REPORT X-073-74-42" GSFC, JAN 1974	"SEASAT TASK TEAM REPORT" "SEASAT SOURCE VERIFICATION STUDY," GE FINAL REPORT, 4/8/74	"SSPD DATA SHEETS OP-07-A" REV. 10/15/73

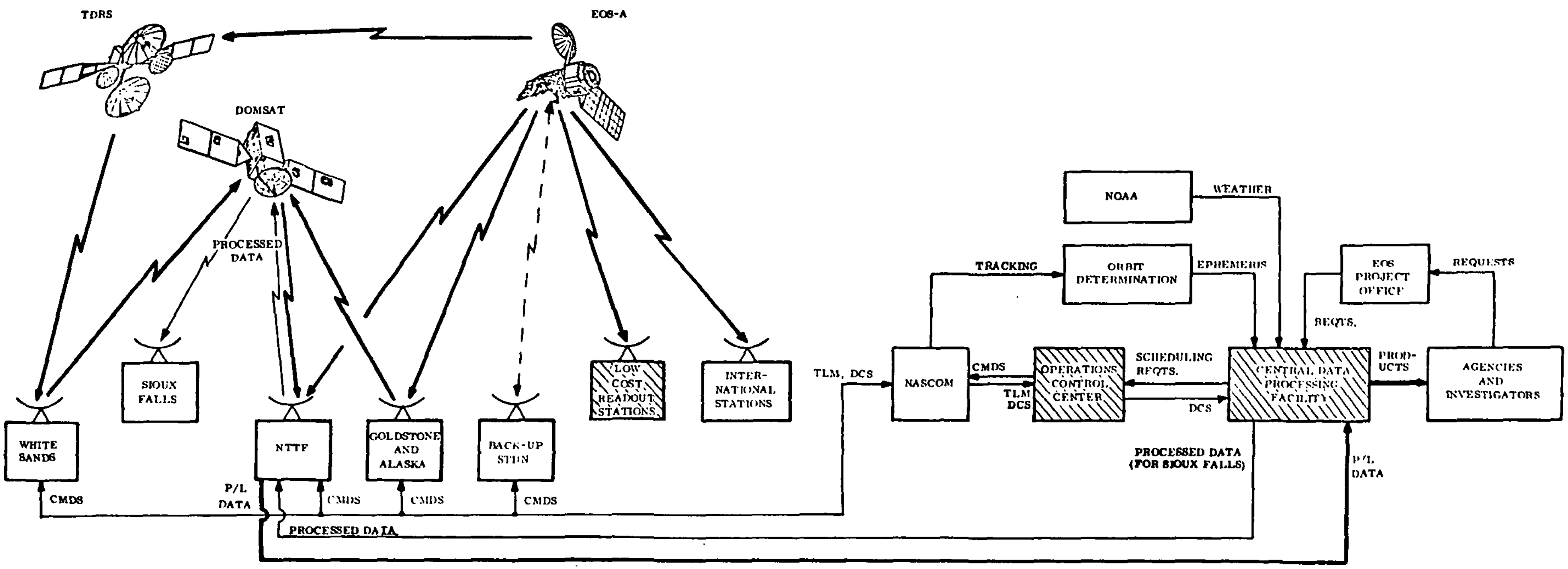


Figure 6-1. EOS System Configuration

stations or through the spacecraft 8-foot dish to TDRSS. Limited commanding, at reduced data rate, is possible from TDRS through the spacecraft omni antenna.

The Operations Control Center (OCC) is the focal point of all mission orbital operations. Here the overall system is scheduled, spacecraft commands are originated, and orbital operations are monitored and evaluated. Telemetry and command data transfer between the OCC and remote ground sites is accomplished by NASA Communications (NASCOM).

The Central Data Processing Facility (CDPF) accepts payload data in the form of magnetic tapes recorded from direct transmission to the NTTF station or by DOMSAT relay. The CDPF then performs the required correction and annotation of the data and prepares master high density digital tapes of all data processed. Output products for users in the form of computer compatible tapes and color and black-and-white imagery may be prepared off-line using these master tapes. The CDPF includes a storage and retrieval system for all data and provides for the delivery of data products and services to investigators and other data users.

6.3 EOS-A BASELINE SPACECRAFT

The Baseline EOS spacecraft has been configured for launch by the 2910 Delta booster using the standard eight-foot diameter fairing, and has the capability for retrieval by Shuttle. The orbital configuration is shown in Figure 6-2 along with key subsystem design features.

The EOS-A mission payload consists of the five-band MSS and six-band Thematic Mapper instruments. An eight-foot deployable gimballed antenna is provided for direct wideband payload communication with TDRSS spacecraft. Two X-band antennas are provided for direct communications with the STDN stations.

The EOS modular spacecraft design as illustrated on Figure 6-3 has an aft Subsystem or "Bus" section and a forward instrument section. The Bus section consists of a core structure supporting Attitude Control (ACS), Power, Communications and Data Handling

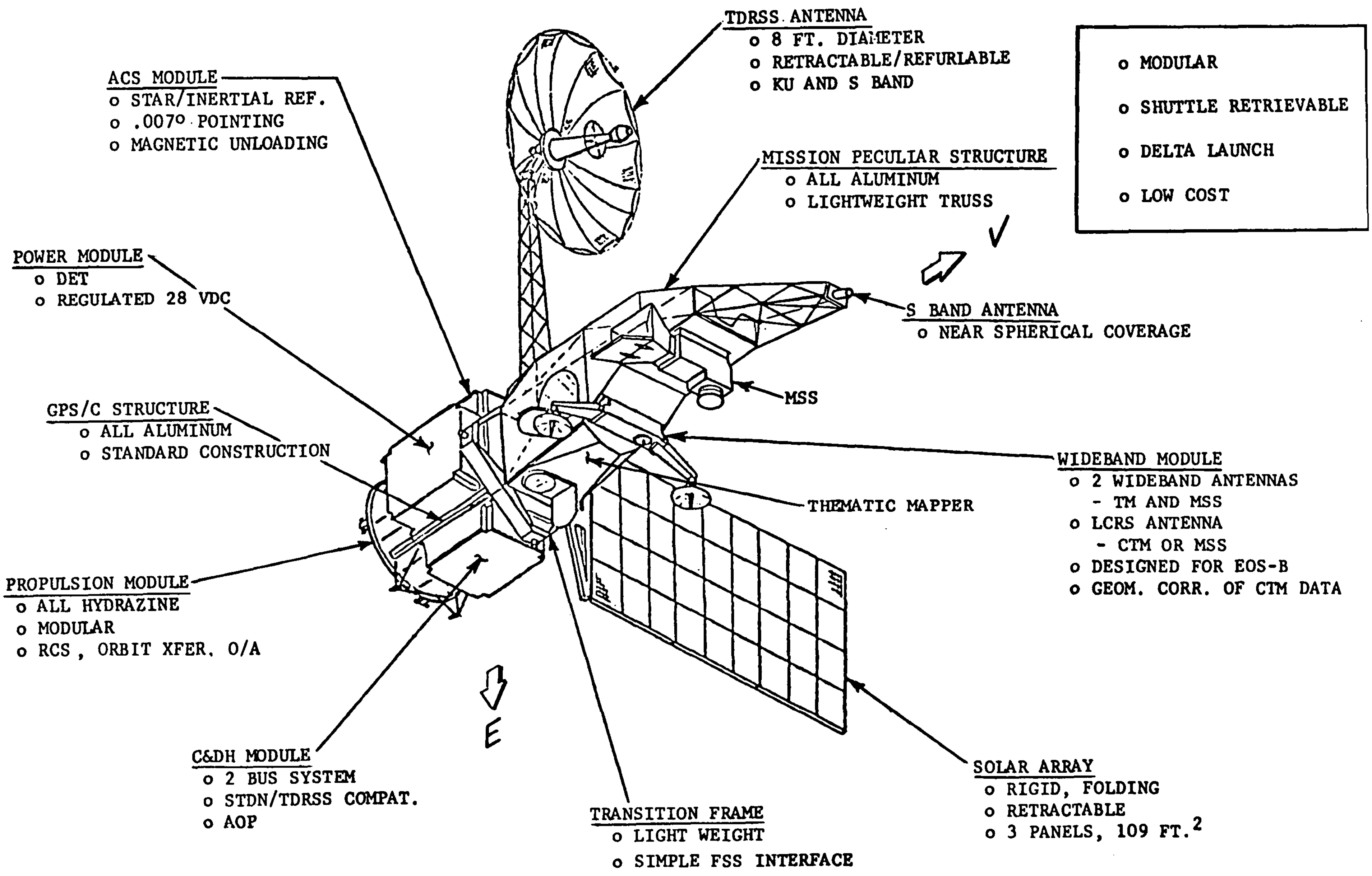


Figure 6-2. EOS-A Spacecraft Orbital Configuration

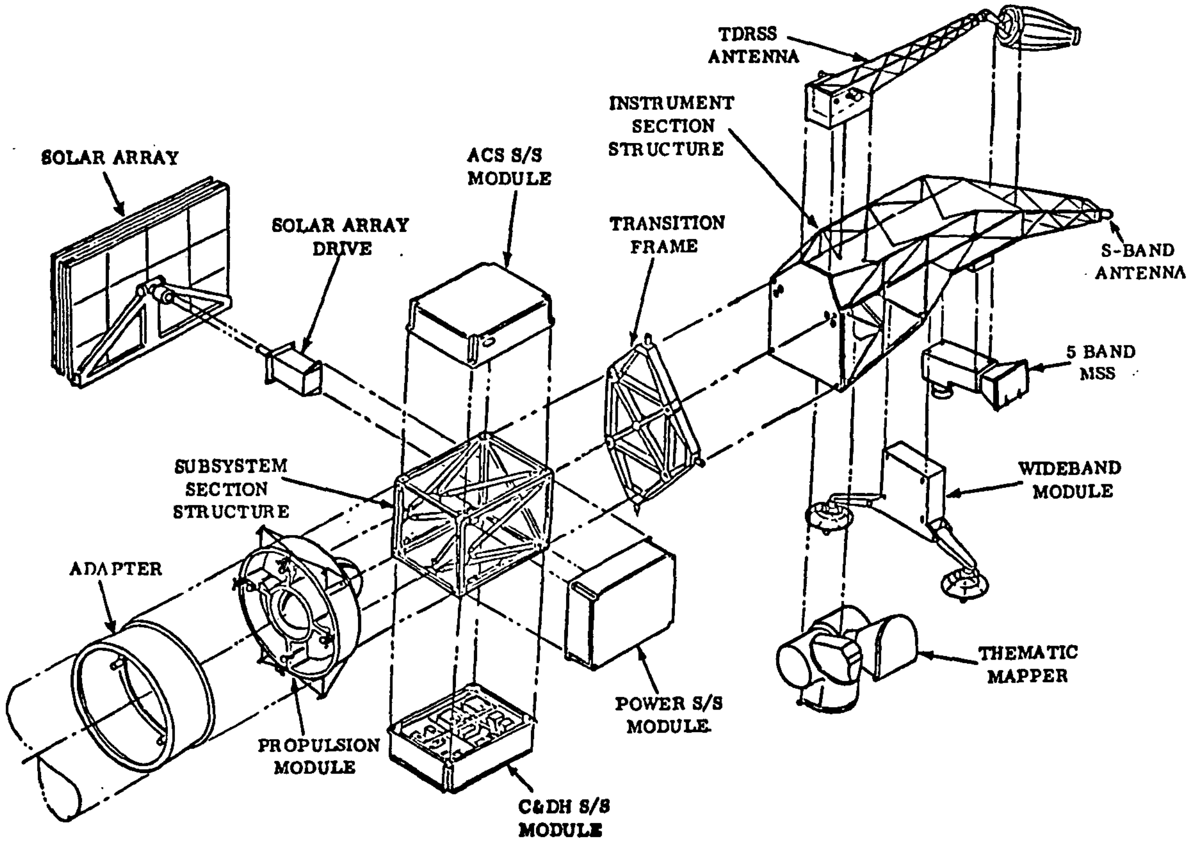


Figure 6-3. EOS-A, Exploded View

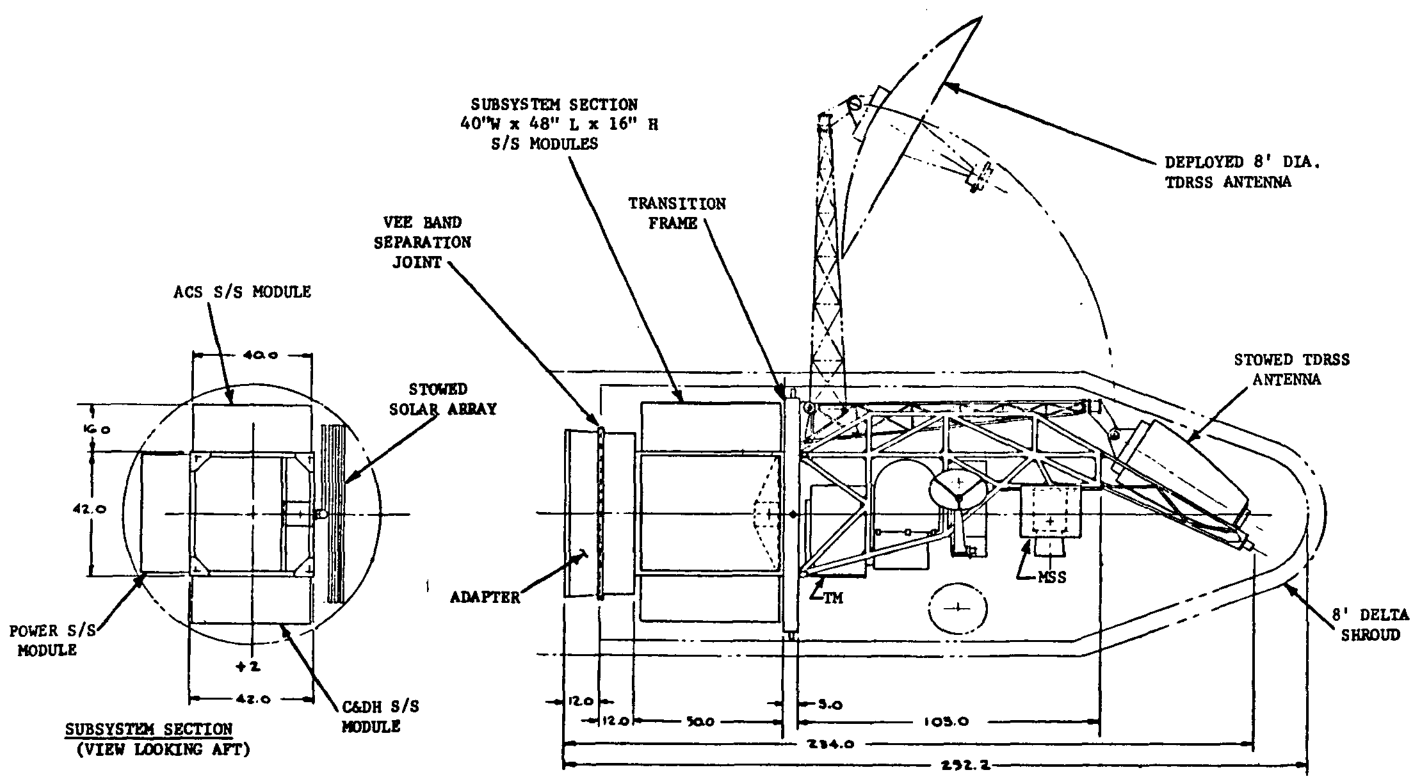


Figure 6-4. Launch Configuration

(C&DH) subsystem modules, the Propulsion Module, and the Solar Array Drive. The aft end of the Propulsion Module is attached to a conventional conical adapter via a Vee-band separation joint for Delta or Titan launch. A transition frame is located between the Subsystem and Instrument Sections for three-point attachment to the Shuttle for launch or retrieval. The folded solar array is stowed on the Spacecraft side opposite the Power Subsystem module. The design has outstanding design flexibility to accommodate alternate missions.

The Spacecraft launch configuration is shown on Figure 6-4 for the Delta launch vehicle. The Delta fairing imposes the most severe space constraints and has dictated the overall spacecraft geometry and the deployment requirements for the array, TDRS antenna, and X-band antennas.

The Subsystem Modules and the folded solar array form a central cavity housing the propulsion tanks and solar array drive. This rectangular arrangement was selected for the Subsystem Section to provide maximum space utilization within the 86-inch diameter Delta shroud envelope. Subsystem modules, sized to fit this arrangement, are 40" W x 16" H x 48" L. This module size contains all subsystem components and includes adequate growth capability for advanced missions.

The Instrument Section arrangement positions the TM and MSS instruments to provide a clear field to view toward earth for sensor apertures and toward space on the anti-sun side for the instrument radiation coolers. The wideband module is positioned between the TM and MSS to provide a clear field of view for the deployed antennas. Two 1.7 foot diameter, two axis gimballed deployable antennas, and a single fixed Low Cost User antenna are provided for wideband communications. An eight-foot diameter furlable antenna mounted to a two-axis gimbal drive and deployable boom is provided for TDRSS, and is stowed above the instruments.

The solar array drive is mounted to the forward end of the Subsystem Section and the array is folded alongside the Subsystem and Instrument sections. This stowage arrangement

results in a wider, shorter, deployed array with adequate growth capability for advanced missions requiring a higher output array. For Shuttle launch or retrieval the spacecraft is supported at the Transition Frame separating the Subsystem and Instrument sections. Note that the Spacecraft has not been designed for resupply but does include provisions for launch or retrieval by Shuttle. The basic modular design, however, can be adapted for resupply by the addition of resupply latches and remateable electrical interface connectors. These provisions were not incorporated in the Delta launched EOS-A configuration because of the excessive weight penalty (approximately 400 pounds).

A weight summary for the EOS-A spacecraft is shown in Table 6-4.

6.3.1 STRUCTURE

The EOS structural arrangement shown on Figure 6-5 uses a conventional conical adapter rigidly attached to the booster interface and attached to the spacecraft by a circumferential vee-band separation joint. The subsystem support structure is an aluminum truss attached to the forward face of the built-up cylindrical propulsion module at eight points. The propulsion section redistributes loads from these eight hard points to the vee-band joint. A Transition Frame, attached at the forward corners of the box truss and separating the Subsystem and Instrument Sections, provides a three-point retention interface for Shuttle launch or retrieval.

Subsystem modules are mounted to the upper, lower, and anti-sun sides of the box truss and the solar array drive is attached internally in the forward area aft of the Transition Frame. This Subsystem Section, composed of box truss, subsystem, and propulsion module shell, forms the basic Bus common to all EOS configurations.

The forward Mission Peculiar instrument support structure is attached to four corner fittings on the Transition Frame, and all loads are carried through the subsystem box truss and propulsion module shell structure to the adapter and booster interface for a Delta or Titan launch. For a Shuttle launch or retrieval, the spacecraft is retained at the central transition frame.

Table 6-4. EOS Weight Breakdown (Pounds)

Basic Spacecraft		(1115)
Structure & Modules	360	
Attitude Control	90	
Power	222	
Communications & Data Handling	184	
Harness & Signal Conditioning	110	
Thermal	38	
Pneumatics	40	
Adapter	71	
Total Mission Peculiar		(762)
Structure	185	
Solar Array & Drive	114	
Harness & P/L Remotes	35	
Thermal	29	
Orbit Adjust	45	
Orbit Transfer	145	
Wideband Comm.	134	
TDRSS	75	
Payload		(505)
Thematic Mapper	350	
MSS	155	
Weight Contingency		(200)
TOTAL SPACECRAFT		2582 (1171 Kg)

This conventional arrangement has been selected for EOS for its significantly lower weight, providing maximum payload weight capability and margin, and for its simplified vee-band separation system.

Arrangement and construction of the ACS, Power, and C&DH modules is illustrated on Figure 6-6 for a typical subsystem module. These modules are designed to reject all excess heat outboard with all side and inboard surfaces covered with multi-layer insulation blankets. Components are mounted directly to the inner face of the one-inch thick aluminum honeycomb sandwich outer panel. The outer panel is integrally stiffened by keels

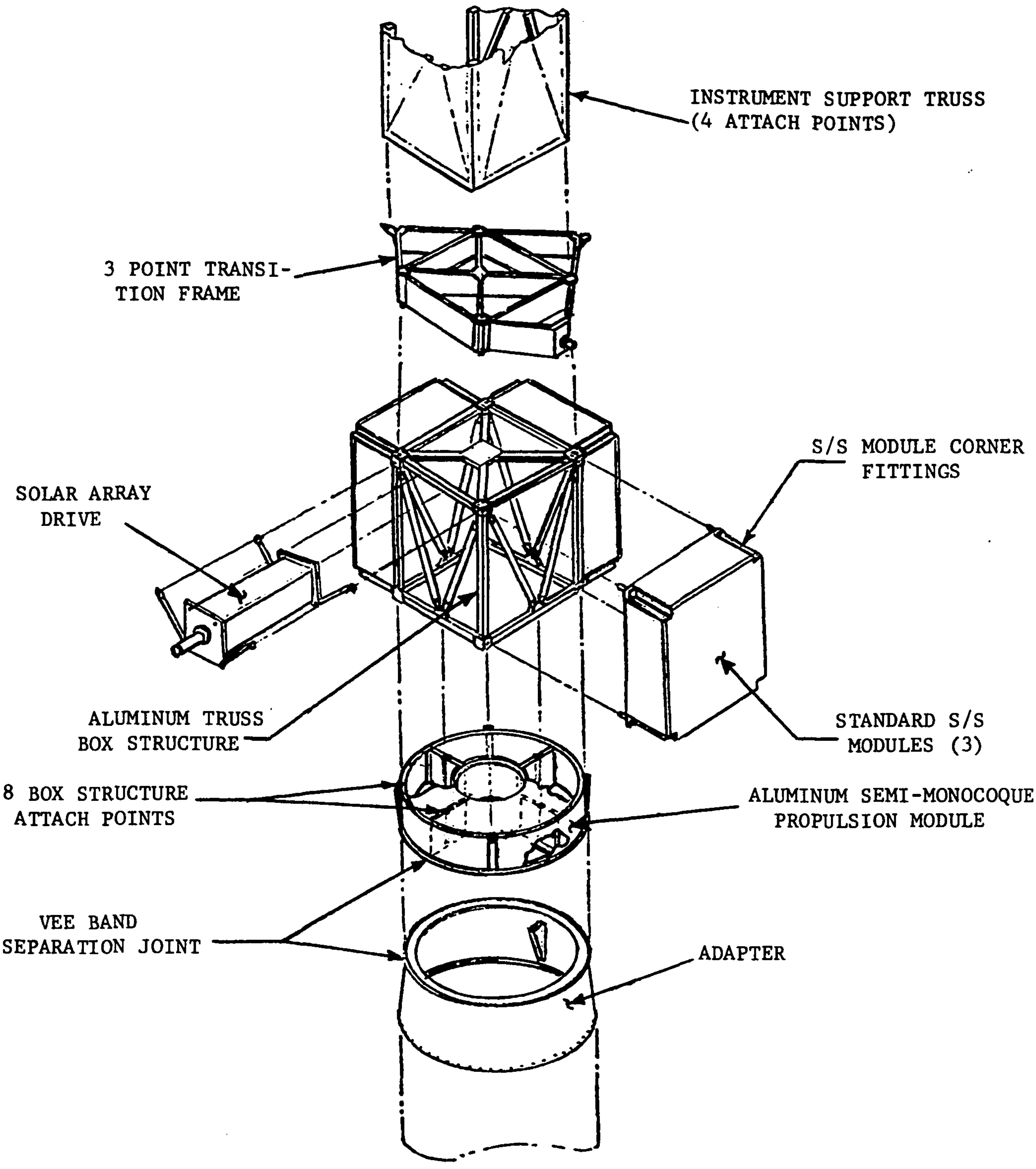


Figure 6-5. EOS Structural Arrangement

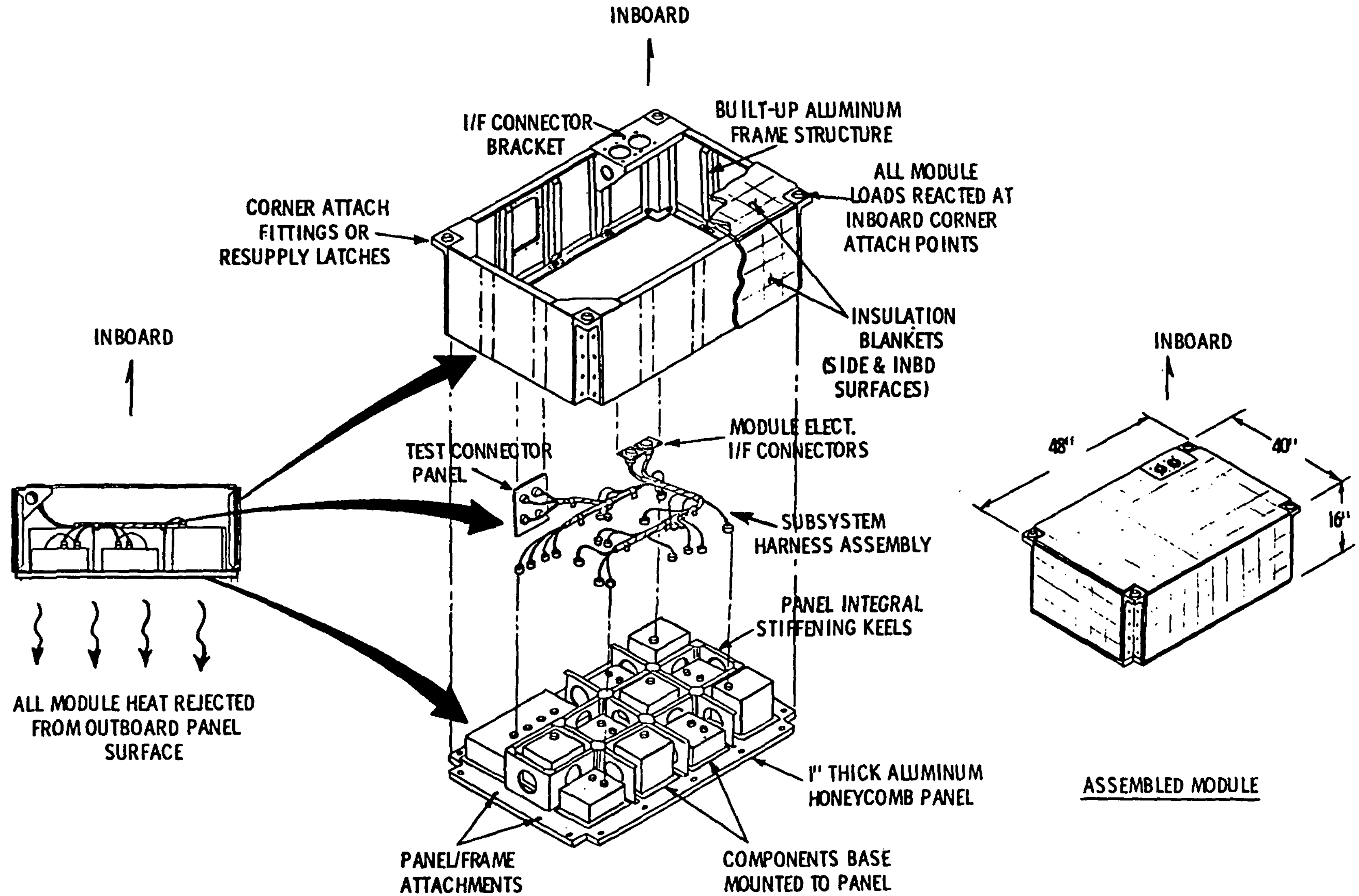


Figure 6-6. Subsystem Module Arrangement

tailored to the individual component arrangements. A subsystem harness interconnecting the components and interface and test connectors is designed for fabrication and installation as a unit. Once the harness is installed and clamped to the keel the module may be bench tested prior to installation of the frame structure and insulation covers. This "breadboard" subsystem assembly on the outer panel provides maximum ease of installation and replacement of components during the assembly cycle.

6.3.2 ATTITUDE CONTROL

The EOS Attitude Control Subsystem consists of nine major components (Figure 6-7); two of which, the On-board Computer (OBC) and the Propulsion Reaction Control Subsystem, are physically located in separate modules. The principal attitude sensing component is the Fixed Head Star Sensor, which periodically updates the ACS attitude. With star updates every 1000 seconds or less, the ACS is capable of providing pitch and roll accuracies of .006 degrees, and yaw accuracies of .002 degrees. This accuracy is maintained between star updates by the Inertial Reference Unit acting in concert with a Kalman Filter (an on-board computer routine) which continuously aligns and calibrates the IRU. The use of this filter significantly reduces the IRU and Star Sensor requirements, and permits the selection of inexpensive off-the-shelf components (Kearfott double degree of freedom gyros and the Ball Brothers Star Tracker). The noise characteristics of the IRU, including the effect of sample data noise and momentum wheel "pulsing," are low enough to keep the spacecraft jitter amplitude below .0003 degrees at all frequencies above 2×10^{-4} rad/sec; a value well below those specified for the ACS.

The Sperry momentum wheels selected for the ACS have sufficient momentum storage capability (7 lb-ft-sec) and torque (23 oz-in) to satisfy the EOS missions, as well as the SEOS and Solar Maximum Mission, yet are weight competitive with wheels of lesser momentum capability. At low altitudes the momentum wheels are unloaded by magnetic torquers with a capability of 30,000 pole-cm per axis. The earth's magnetic field is estimated by the computer, eliminating the need for a magnetometer.

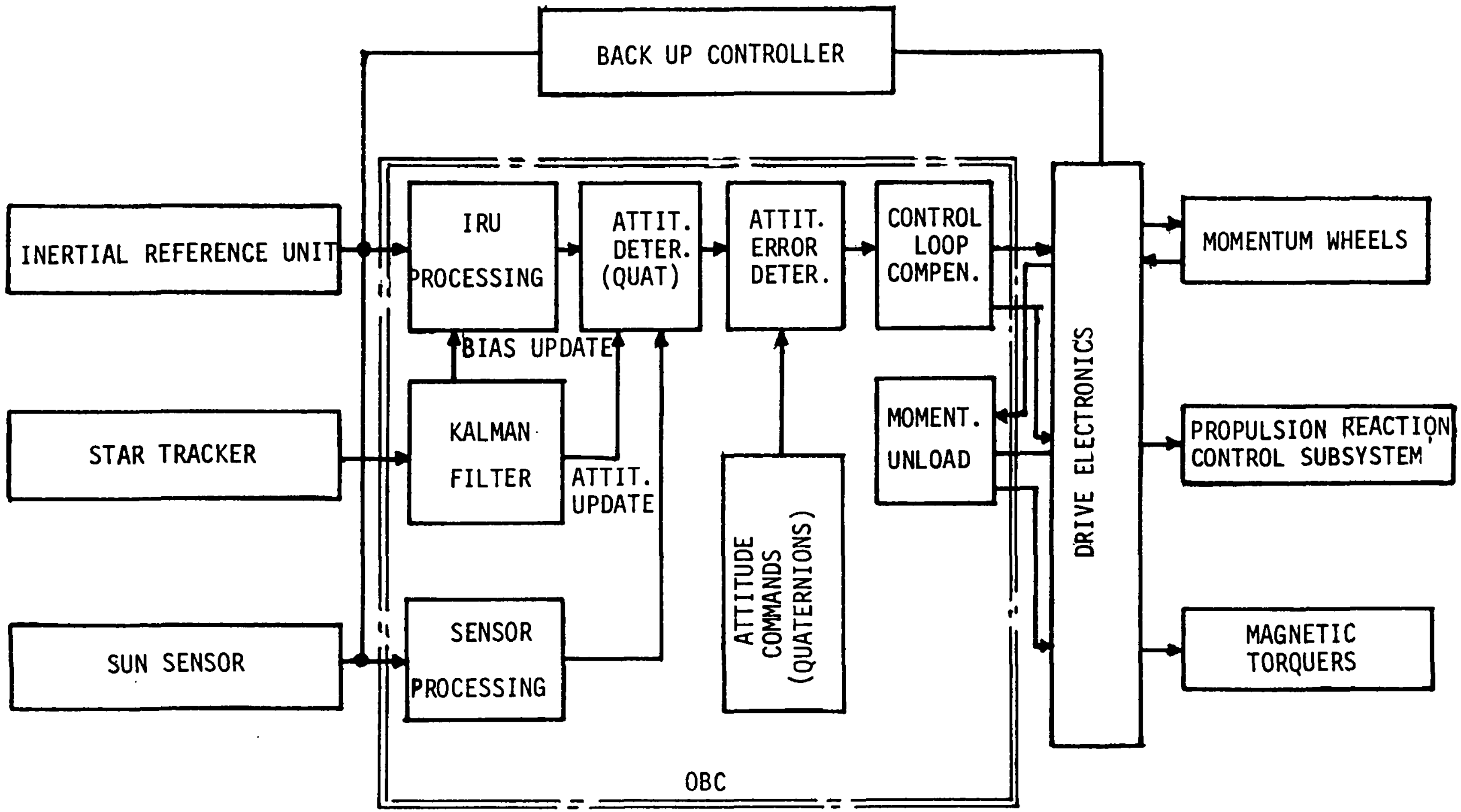


Figure 6-7. Simplified ACS Block Diagram

The Propulsion Reaction Control Subsystem is used for initial acquisition, backup momentum unloading, and for low earth orbiting missions. Acquisition is accomplished with the normal ACS assisted by an off-the-shelf solar aspect sensor (as a coarse sensor) and special on-board software. Once acquisition has been completed, the spacecraft can be commanded to provide earth coverage or point a payload sensor (such as sun sensor or star telescope) with accuracies well within the field-of-view of most payload sensors (two arc minutes after a 90-degree slew). The ACS is capable of performing any of the defined missions with no change to the components and little change to the software. The use of quaternions (Euler parameters) to specify the spacecraft orientation provides the flexibility to operate in either earth oriented or inertial missions, and automatically defines the logic necessary to execute large and small angle maneuvers or corrections on a three-axis basis.

The ACS, excluding the On-board Computer and the Propulsion Reaction Subsystem, is packaged in the standard 48-inch by 40-inch by 16-inch module.

6.3.3 POWER SUBSYSTEM

The power subsystem uses a Direct Energy Transfer (DET) implementation which provides a highly regulated bus (+28 ±0.3vdc) for distribution to the user subsystems and experiments. A modular approach for the battery charge/discharge electronics is used to provide mission flexibility.

A simplified functional block diagram of the subsystem is shown in Figure 6-8. The Central Control Unit senses the bus voltage level and generates a control voltage based on the detected error. This control voltage is used to control the operation of the battery discharge boost converters, battery charge controllers, and sequenced partial shunt regulator. The operation of these components is such that the load bus is automatically provided with first priority to the solar array power at all times. Battery charging has second priority, as modified by the other control inputs to the battery charge controllers, with excess solar array power automatically dissipated in the sequenced partial shunt regulator.

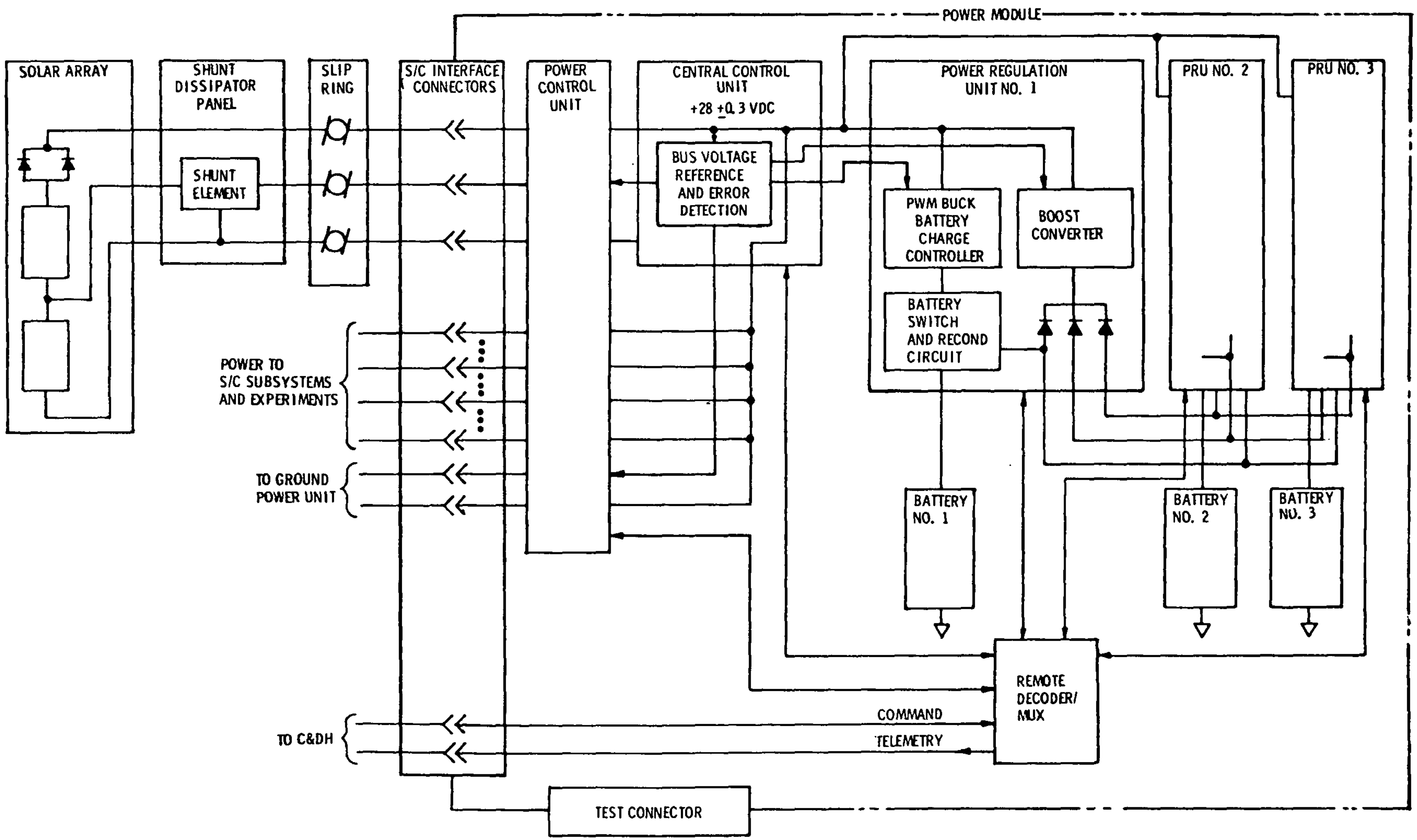


Figure 6-8. Power Subsystem Baseline Design

The Power Regulation Unit (PRU) contains the charge/discharge electronics which are associated with each battery. The basic spacecraft Power Module contains three PRU's, one associated with each battery. Each PRU contains a PWM buck battery charge controller, which is dedicated to one battery, and a PWM boost converter which receives discharge current from all batteries in the subsystem. Each individual boost converter has an output rating of 450 watts. The PRU also contains the battery discharge isolation diodes, charge disable relay, and battery reconditioning circuitry (if required). The PWM buck battery charge controllers provide charge current limiting and voltage limiting at one of eight ground commandable, temperature compensated levels.

The power subsystem has been designed with adequate internal redundancy to provide a high probability of achieving the two-year mission design life time. Majority voting, quad redundant logic is used in the Central Control Unit to provide the necessary reliability associated with the generation of the regulation control voltage. The energy storage has been sized to enable nearly full EOS-A experiment operation with one battery failure. The failure of one boost converter will not limit the full operation of the EOS-A payload.

The solar array design uses a modular construction approach to permit easy growth in the array capability for a variety of missions. The basic array building block unit is a 60 by 21.75-inch subpanel consisting of a 0.25-inch thick honeycomb substrate on which three solar cell circuits are mounted. Each circuit consists of a matrix of 308 20x40 mm cells which are connected 77 in series by 4 in parallel. Four of these standardized subpanels are mounted on a built-up aluminum frame structure to form a solar array panel. Three such panels are hinged together to form the complete solar array assembly for EOS-A.

6.3.4 C&DH MODULE

The C&DH Module contains eight major components necessary to provide spacecraft tracking, ground and on-board control of all spacecraft and payload functions, retrieval of narrowband and mediumband ($\leq$ 650 kHz) observatory data, and a coherent clock and timecode signal for use by spacecraft subsystems. The subsystem block diagram is given in Figure 6-9.

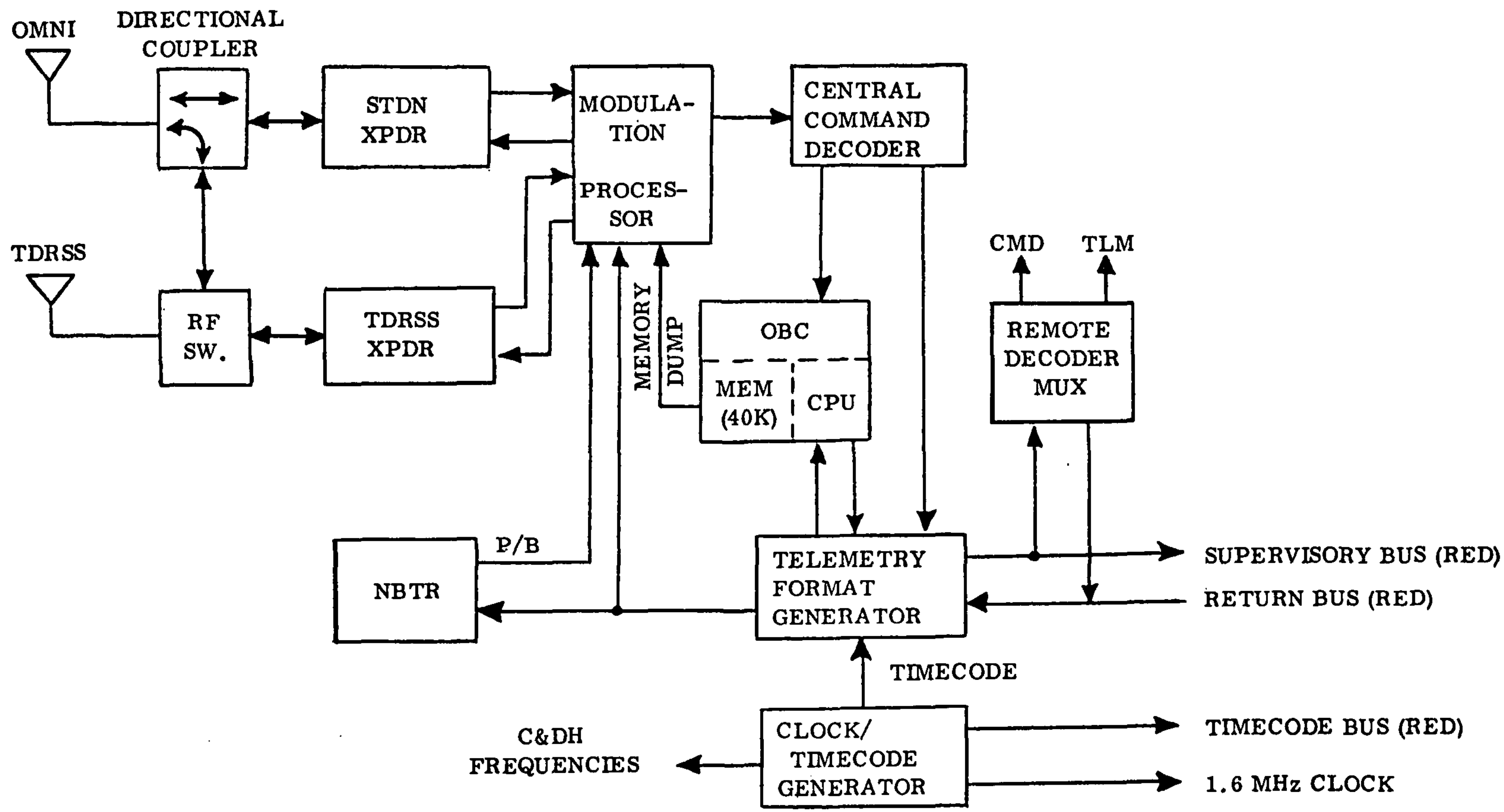


Figure 6-9. C&DH Subsystem

Tracking, telemetry, and command/control are provided through two independent S-band links: one compatible with STDN; one compatible with TDRSS. The STDN link operates into a Motorola SUB transponder and is used for GRARR and transmission of up to 50 commands per second. The TDRSS link operates into a Magnavox transponder and uses a PN code for ranging and transmission of up to 25 commands per second. Narrowband telemetry data may be transmitted over either link at 1, 2, 4, 8, or 16 kbps (selectable by command); mediumband data may be transmitted at 650 kbps over STDN and 560 kbps over TDRSS.

Uplink command data are processed by a modulation processor similar to the ERTS PMP and decoded by a central command decoder. These data may be used to reprogram the digital on-board computer (OBC) or to execute (real-time or delayed) any of 2048 pulse or 128 serial magnitude (16 bit) commands. Commands can also be executed as necessary by software in the OBC. A telemetry format generator controls the sampling of 2048 different telemetry functions (analog, bi-level digital, and 512 serial digital) and formats the data into a 128 x n ($n \leq 128$, binary) frame of 8 bit words for transmission to the ground and the OBC. These functions can also be interrogated by the OBC. Both command and telemetry data are handled by a dual digital data bus system which interrogates and receives data from remote decoder/muxes located in the user subsystems. The clock and timecode generator issues a standard 1.6 MHz clock signal as a coherent timing source for use by spacecraft subsystems. A one millisecond timecode (incremental counter) is also provided for annotation of data. This timecode is reset to zero at the beginning of each month.

The OBC is the NASA/GSFC AOP and is used for computational support of the attitude control subsystem through the command and telemetry data busses. It also is used to control spacecraft operation and assess spacecraft health. A total of five memory modules are used to provide 40K words of memory.

The narrowband tape recorder is the NASA/GSFC universal 10^9 digital recorder and will be used to provide back orbit data for diagnostic use on early missions.

6.3.5 ELECTRICAL INTEGRATION

The electrical integration subsystem consists of all intramodule harnessing and selected electronics not included in the three basic spacecraft modules. A block diagram of the spacecraft electrical system is given in Figure 6-10. The first part of this diagram shows the three basic modules comprising the spacecraft bus, along with the reaction control system, the solar array, and the signal conditioning and control module (SCCM). The second part shows the mission peculiar equipment.

All intramodule harnessing is separated by function (i.e., power, command and data busses, timecode and clock frequencies) and wrapped with copper tape shielding to minimize EMI. Shields are tied to the chassis of the user subsystem for all signals less than 100 kHz, except for cables carrying currents in excess of 5 amps for periods less than 100 msec (pyro and solenoid drives) which have the external shield tied at both ends. Signals in excess of 100 kHz also have shields tied to chassis at both ends.

All components within each spacecraft module have their cases electrically tied to the module which is, in turn, electrically tied to the spacecraft frame. All components, with the exception of RF devices, provide isolation between power and signal grounds by means of a DC/DC converter in the power input circuit. All power grounds (primary return of DC/DC converter) are tied to a single power return in the module and then returned to the spacecraft power ground in the power module, which is solidly tied to the spacecraft unipoint ground on the transition frame. All signal grounds within a module are tied to a single power return in the module and then returned directly to the spacecraft unipoint ground.

The signal conditioning and control module (SCCM) contains a number of standard circuits applicable to all missions and several unique to a given mission. The former include structure heater control, structure thermistor signal conditioning, solar array drive control, adapter separation, solenoid drivers, and pyro drivers. The mission-unique circuitry for EOS-A includes the orbit transfer solenoid drivers, deployment of the solar array, deployment of the TDRSS antenna, unlatch for the STDN antennas, and shuttle caution and warning circuitry.

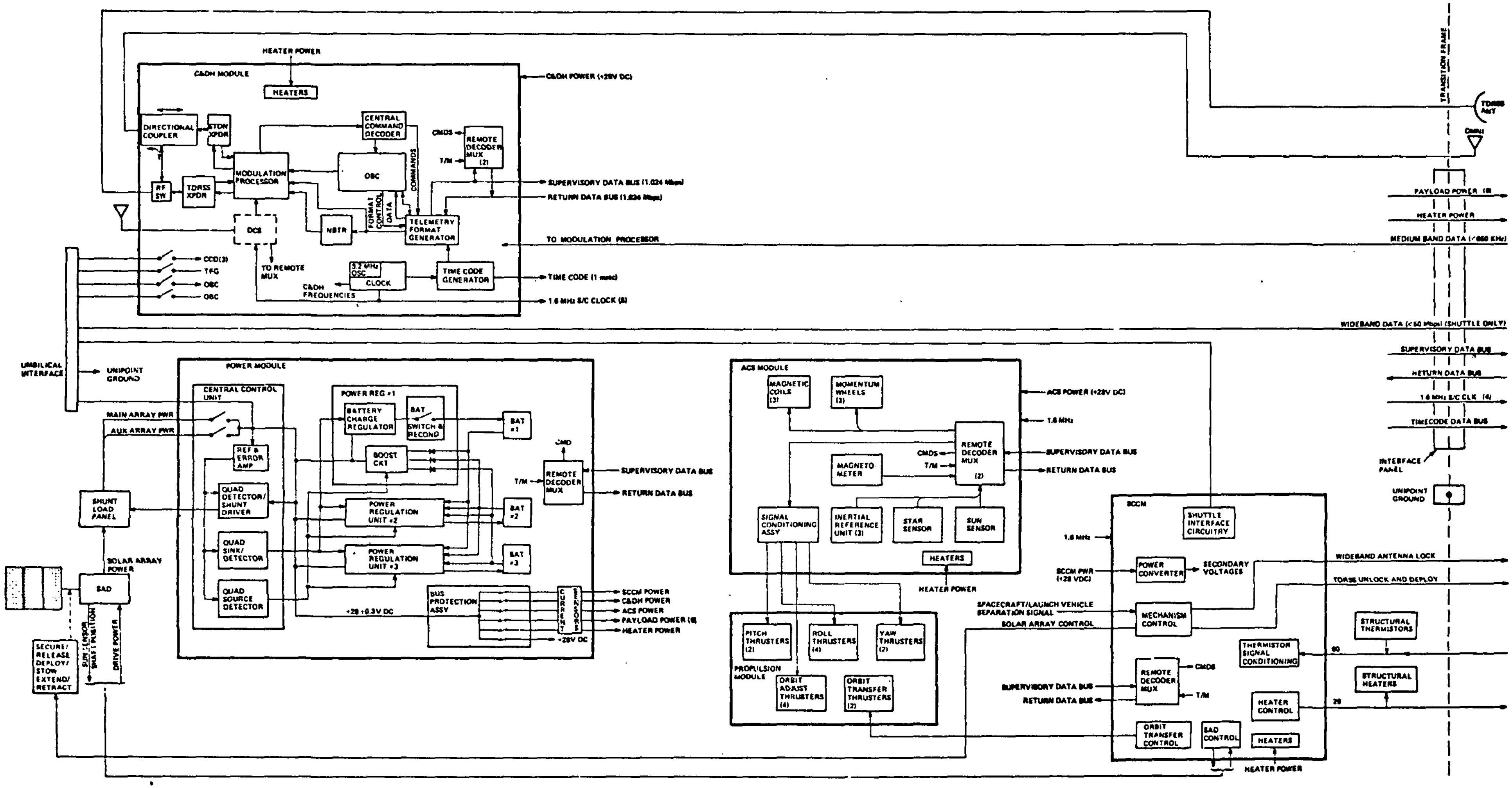


Figure 6-10a. Electrical System Diagram (Basic Spacecraft)

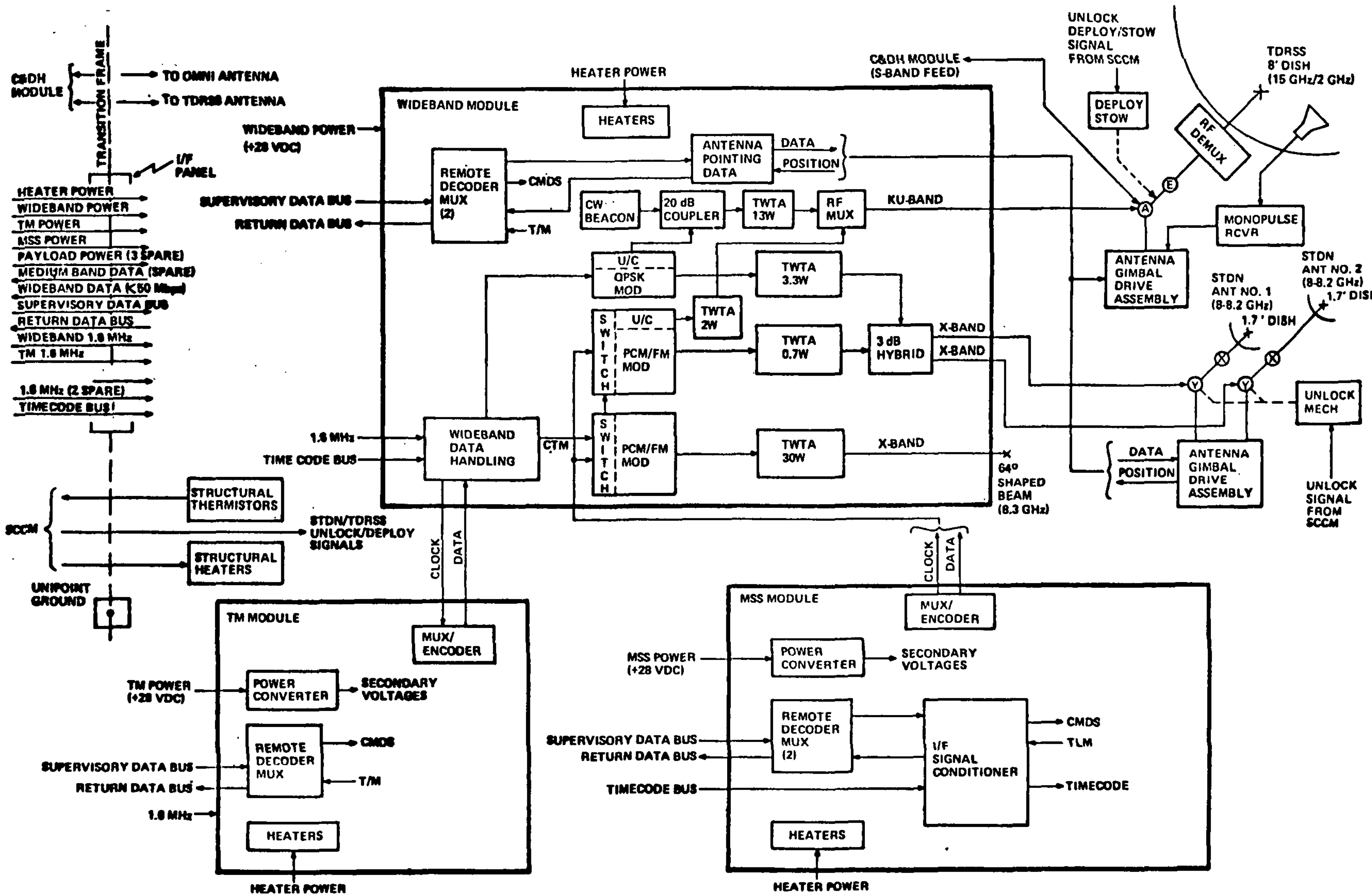


Figure 6-10b. Electrical System Diagram (Mission Peculiar)

6.3.6 THERMAL CONTROL

The thermal control subsystem maintains all vehicle temperatures and temperature gradients using a simple, reliable, control concept with thermal insulation and coatings supplemented by electronic thermostat and command activated electrical heaters. The thermal control subsystem passive elements include multi-layer insulation blankets, thermal coatings, conduction spacers, and thermal grease. Active elements include command or electronic thermostat activated electrical heaters.

The design approach for the subsystem modules utilizes the thermal coating optical properties to control the amount of energy which is absorbed from external vehicle fluxes and rejected from external vehicle radiation surfaces. This radiation area is sized to reject the absorbed external heat fluxes and maximum orbit average internal heat dissipations while maintaining the maximum average temperature specified, usually $70 \pm 5^{\circ}$ F. Five mil teflon over silver thermal control coating has been selected for the radiator areas.

The propulsion module thermal concept is also passive with thermal insulation and coatings. Local electronic thermostat activated heaters are used at the locations where the engines protrude the insulation.

The solar array drive is insulated from the external environment and structurally hard mounted so that the structure provides a thermal sink which maintains the array drive within temperature limits, since the small array drive power is easily dissipated by leakage at the shaft exit areas. The solar array rear surface is coated with S-136 white paint to minimize the external effect of earth albedo at low altitudes.

The instrument module support structure is completely enclosed in multilayer insulation. The outside surface of the insulation will have a thermal control coating with an α/ϵ ratio selected to maintain the average structural temperature just below the required nominal temperature for the instruments. The instruments will be isolated from the structure at the mounting interfaces using conduction isolation spacers. These spacers will maintain near-adiabatic conditions between the instruments and structure, limiting the average

heat exchange to acceptable values (nominally less than 10% of the instrument dissipation). The structure will be black anodized to maximize internal radiation exchange and limit structural temperature gradients. The instrument modules will be independent thermally. They will be completely enclosed in multi-layer insulation except for attachment points, apertures, cooler protrusions, and heat rejection surfaces.

6.3.7 PROPULSION

The propulsion subsystem provides the three functions of reaction control, orbit adjust, and orbit transfer. The primary reaction control capability is for initial attitude acquisition or reacquisition. Backup capability is provided for momentum wheel unloading and limit-cycle attitude control. For orbit transfer and orbit adjust, capability is provided for (a) removal of launch vehicle injection errors, both in plane and cross track; (b) orbit maintenance at the desired mission altitude (775 km); (c) orbit transfer to Shuttle retrieval altitude (612 km); and (d) maintenance of spacecraft attitude control during these orbit adjust and orbit transfer maneuvers.

A mass expulsion, monopropellant, hydrazine fueled propulsion system was selected as the most cost effective and flexible propulsion type for either Delta or Titan launched spacecraft. The functional block diagram is shown in Figure 6-11. The propellant and pressurant is stored within a cylindrical pressure vessel using a surface tension type propellant management device which retains the propellant at the tank outlet port for full time availability to the engines under all on-orbit operating conditions. The tank has a manually operated fill and vent valve on the pressurant side and a fill and drain valve on the propellant side. A pressure transducer is located in the outlet line from the propellant tank, the output of which can be monitored periodically via telemetry as a "health check" of the system and to determine the quantity of propellant available. Propellant flowing from the tank is filtered through a high-capacity, low-micron-rating etched disc filter. The filter is located upstream of the isolation valve and the engines in order to provide adequate contamination protection. Propellant isolation valves of the latching type are located in the propellant feed lines to isolate the propellant tank from the engine thruster groups during long periods of non-usage. Downstream of the latching valve, distribution

piping is used to feed propellant to each of the rocket engine assemblies. Each assembly consists of a solenoid operated propellant control valve and a thrust chamber. The thrust chamber consists of a propellant injector, a spontaneous catalyst (SHELL 405) and a converging-diverging conical nozzle. Operation of the solenoid valves is by an electrical command.

The reaction control thrusters are positioned in bow-tie configuration at four locations near the aft end of the EOS-A spacecraft. This configuration provides three axis motion of the spacecraft using a minimum number (eight) of thrusters. One orbit adjust thruster is also positioned at each of these four locations but is oriented such that the nozzles point in the aft direction. This configuration permits orbit adjustments using opposite pairs of engines or reaction control thrusters in conjunction with a single orbit adjust engine. Redundant orbit transfer engines are located in an aft pointing orientation and provide for orbit transfer prior to spacecraft retrieval.

The packaging design of the propulsion system is modularized to allow complete assembly and test of the subsystem prior to spacecraft installation.

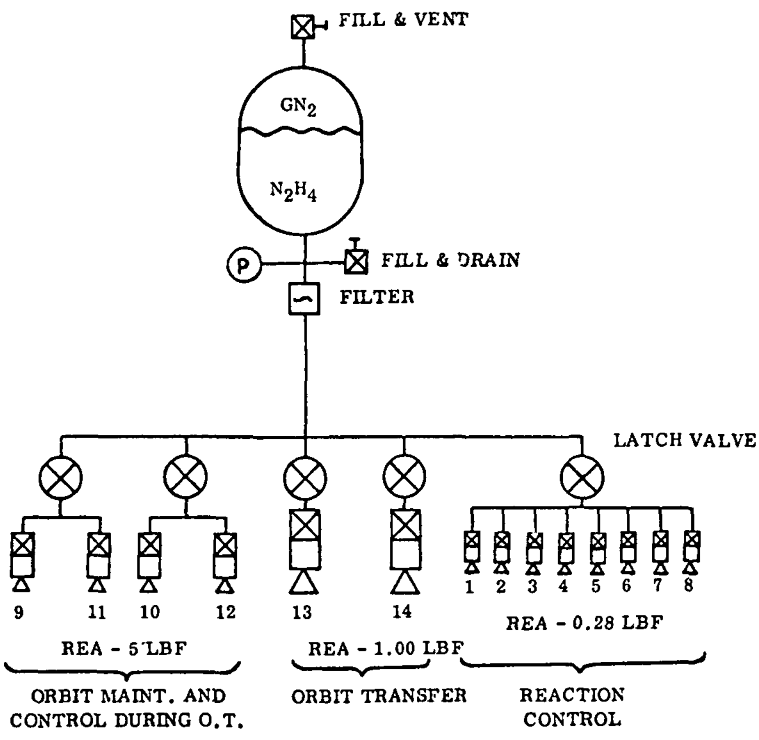


Figure 6-11. EOS-A Propulsion Subsystem Block Diagram

6.3.8 WIDEBAND COMMUNICATIONS AND DATA HANDLING

The Wideband Communications and Data Handling subsystem includes all mission peculiar equipment which interfaces with the MSS and TM wideband data streams and processes and transmits this data to the appropriate receiver(s). The baseline system design provides for transmission of data at Ku-band to TDRS and at X-band to STDN and local user stations.

The wideband configuration is shown in Figure 6-12. Operating data links are as follows:

a. TM and/or MSS with Tracking Beacon via TDRSS. (Compacted TM (CTM) data may replace the MSS data.)

b. TM and/or MSS via the dual STDN links. (CTM data may replace the MSS data.)

c. CTM or MSS data via the local user link.

The Multimegabit Operation Multiplexer System (MOMS) is a high data rate PCM unit consisting of an 84-channel analog multiplexer, an analog-to-digital encoder, and a power converter unit. The multiplexer/encoder modules are mounted as an integral part of the Thematic Mapper in order to minimize input lead length (100 inputs) and reduce the possibility of extraneous noise pickup on the single ended input lines. The converter is separately mounted. The Encoder data output is NRZL serial to the Format Generator.

The Format Generator reclocks the data and performs ancillary/TM data formatting. It contains the logic necessary to generate the preamble, MFS, and 1/4 swath ID words and to insert this information into the data stream. Separate shielded lines are used for the data and the clock signals between the MOMS and Format generator. The signals are both differential to minimize common mode stray pickup. Data signal rise time degradation is removed by the reclocking process.

The Data Buffer interfaces with the OBC, and serves to integrate ancillary data into the wideband format during the "dead time" period following each swath. Telemetry data is buffered and digitally multiplexed by the MOMS. The MF sampling rate, 108,814 words per second, is well in excess of that required for telemetry data.

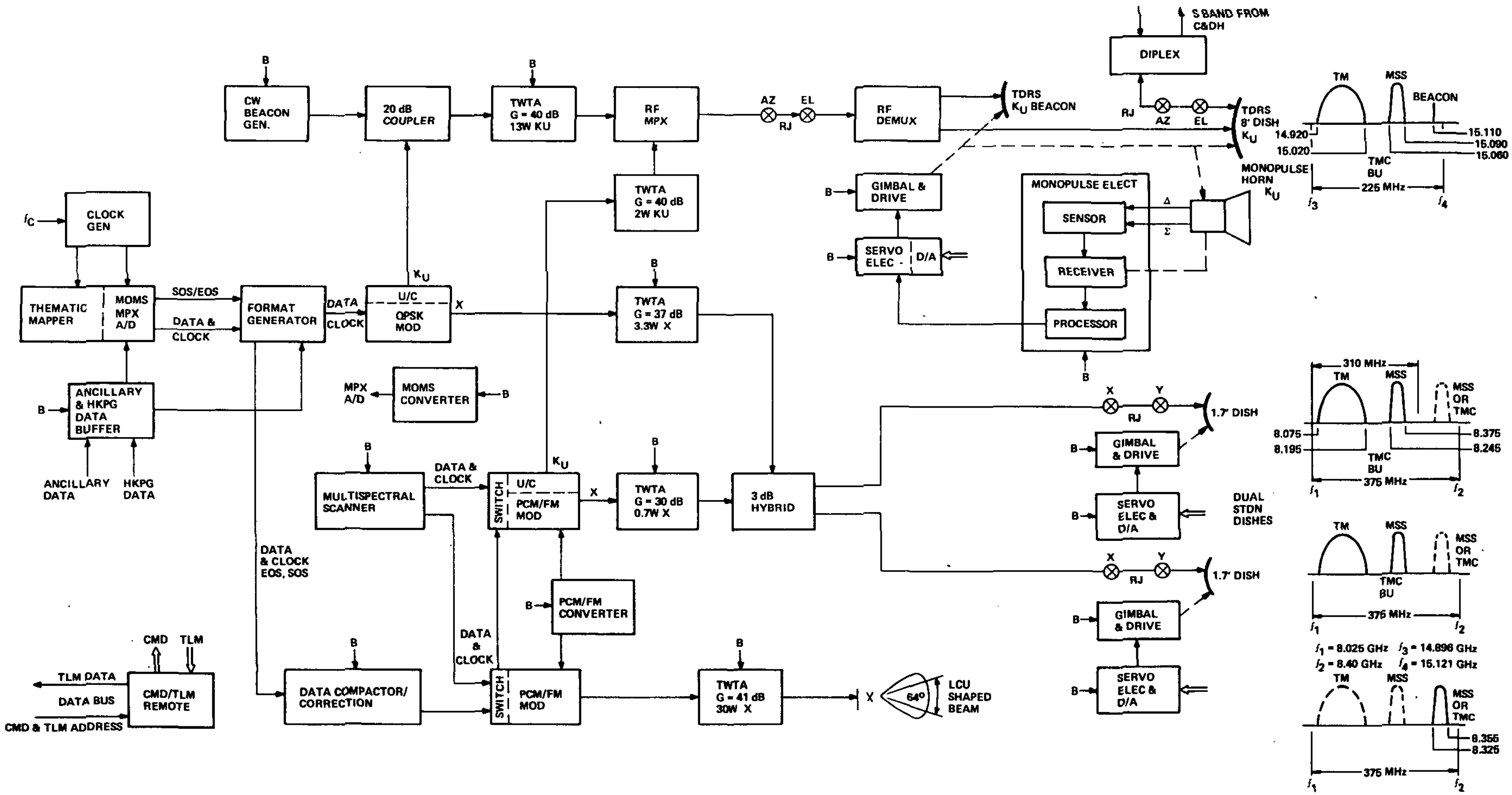


Figure 6-12. EOS-A Wideband Subsystem

The QPSK Modulator accepts the serial NRZL binary data signal and clock from the Format Generator and reclocks, differentially encodes, and QPSK modulates the data. Differential encoding is employed to resolve the four state ambiguity of the QPSK data. Modulation at 400 MHz is employed. The RF spectrum is mixed up to X-band and then mixed up to Ku-band. Parallel modulation at a relatively low frequency is employed since greater design control over modulation performance parameters is possible. Double stage mixing is a convenient means for obtaining simultaneous outputs at both X and Ku-bands. Band limiting filters are used to constrain the TM spectrum to 100 MHz and reduce out of band spurious and crosstalk to the MSS channel. Precise center frequency control is made possible by a reference crystal oscillator.

The Data Compactor/Corrector contains the digital logic, arithmetic circuitry, and speed buffer memory necessary to perform the four modes of TM data compaction. It also implements a geometric x-correction (scan direction) which compensates for earth curvature, the non-linear scan angle, and prescribed instrument scan non-linearities. This provides geometrically accurate (+ 1 pixel) data to the local user.

The PCM/FM modulator is an improved version of the ERTS Wideband Modulator. This version has the AFC loop and reference oscillator deleted resulting in lower size, weight, and power. The open loop stability of the S-band VCO's have a frequency drift less than $\pm$ 1.5 MHz over a one-month period. An additional 10 MHz has been added to the RF spectrum for EOS in order to allow for this drift. Each PCM/FM modulator has redundant commandable VCO's. Simultaneous X and Ku-band outputs are obtained by upconverting the S-band signal. Bandwidth limiting is obtained by premodulation filtering. A switching matrix and data reclocking is also included in the modulator.

Five separate TWT amplifiers are used in the baseline design. Each amplifier is complete with its own high voltage power supply, output isolator (as a protection against inadvertent mismatches), and band pass filter. The band pass filters spectrally limit both broadband output noise as well as the modulation spectrum. Because of the highly non-linear nature of the TWT input/output characteristic and inherent AM/PM conversion,

it is generally not desirable to amplify two separate signals simultaneously since crosstalk will occur. An exception is made in the case of the CW beacon and the Ku-band QPSK modulator signal. Analysis shows that the crosstalk is at an acceptable level in this case, thus eliminating an additional TWT.

In order to limit the number of rotary joints into the TDRS gimbal to a maximum of two (S and Ku-band) the beacon/QPSK spectrum is multiplexed with the PCM/FM spectrum using a directional filter. The combined spectrum is sent through the rotary joint. At the antenna the beacon is stripped off, again by means of a directional filter in the Demux and routed to the TDRS beacon antenna. The TM and MSS signals are routed to the Ku-band feed on the 8-foot dish. Although the Ku beacon dish is shown as a separate antenna, the beacon is actually diplexed into the sum port of the monopulse horn. The horn then serves as the EOS beacon radiator and the receive antenna for the TDRS beacon.

The Monopulse Subsystem acquires the TDRS Ku-band beacon and points the high gain 8-foot dish to TDRSS within 0.1°. The system is a modified version of the monopulse presently being manufactured by General Electric for the Japanese Broadcast Satellite. It consists of a Sensor, Receiver, and Low Frequency Processor. The sensor employs a high performance circular horn antenna utilized as a mode converter which produces a difference pattern similar to other monopulse antennas but requires less space and weight. A time-shared single channel receiver approach is employed in order to minimize relative drift between the delta and sum channels. A ferrite time-share switch and biphase modulator is interposed between the antenna and receiver to condition the incoming RF signal for single-channel processing. A highly selective band pass filter at the input of the receiver protects the system against undesirable signals. Carrier drift is tracked out by a phase locked loop. The low frequency processor separates sum and difference channels, provides low pass filtering and supplies the azimuth and elevation error drive signals to the antenna gimbal.

The main radiator of the TDRS antenna consists of a furlable parabolic reflector. The S/Ku-band coaxial feed is mounted at the focal point of this reflector. The Ku-band

beamwidth is 0.6^{o} and the S-band beamwidth is 1^{o}. Monopulse reception and beacon transmission is achieved via the one-foot parabolic reflector and the monopulse corrugated horn feed located at the focal point. The resulting beam width at Ku-band is 5^{o}. The monopulse electronics is packaged surrounding the horn. A shielding reflective shroud surrounds the one-foot dish.

The baseline wideband system is designed to meet power flux density limitations while maintaining adequate link margin at 10^{-5} BER for all links.

6.4 EOS GROUND SYSTEM

6.4.1 EOS GROUND DATA HANDLING SYSTEM

The EOS Ground Data Handling System (GDHS) is comprised of two major Segments—a Central Data Processing Facility (CDPF) and an Operations Control Center (OCC). The CDPF, in turn, is comprised of a Data Management Element (DME) and an Image Processing Element (IPE). Data flow and interfaces between the EOS GDHS and the other segments of the EOS Ground System, as well as the data flow and interfaces within the EOS GDHS, are shown in Figure 6-13.

6.4.1.1 Data Management Element

The DME provides the centralized management control, monitoring, and reporting of the overall GDHS operations. It provides the focal point for interfacing with the EOS Project Office for requirements, with the NASA Orbit Determination Group for spacecraft orbital definition data, and with the National Oceanographic and Atmospheric Administration for predicted weather information data. The DME is responsible for the generation of payload schedules to satisfy EOS Project Office requirements utilizing the orbital data and weather information, and providing the payload schedule to the OCC. The DME is responsible for receipt and accounting of the raw video instrument tapes, product generation direction to the IPE, and distribution of the output products to the users. The DME also has the management responsibility for monitoring and reporting the overall GDHS performance and spacecraft performance to the EOS Project Office.

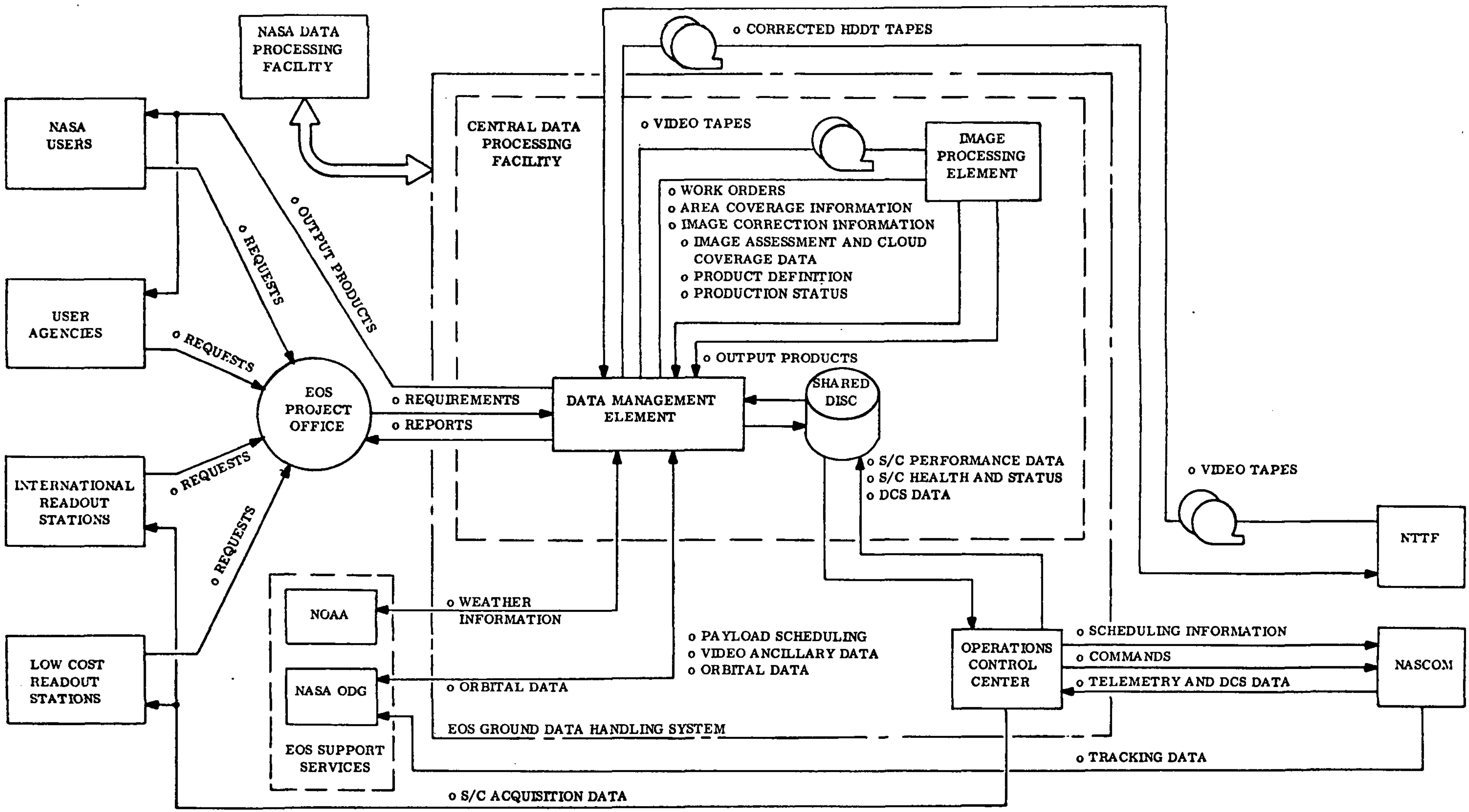


Figure 6-13. EOS Ground Data Handling System

The design approach for the DME is an evolutionary development from the ERTS Data Services Laboratory and contains both new software and hardware to optimize the new and expanded DME requirements for the EOS system.

The DME design approach provides a shared disk for data exchange between the DME and the OCC, computer data communication channels between the DME and the process control mini-computers of the IPE, and computer data communication channels between the DME and outside weather and orbital data interfaces to minimize manual data transfer. The design approach also provides for an expanded work order structure and an integrated product/image data base. The automated production control system includes dynamic work order scheduling with traceability of individual images, work orders, and user requests.

The DME consists of fourteen major software subsystems. Each subsystem is composed of one or more software modules consisting of programs, sub-programs, procedures, and/or overlay segments. In general, the software modules will execute in the environment of the Operating System and Data Base Management System software provided with the central processor. Software subsystems will interface among themselves by mass storage or main memory files unique to one or more subsystems or common to all such as the integrated product/image data base files. The DME computer services hardware subsystem is an integrated set of data processing equipment selected to execute the DME functions under software control. The central processor is characterized by a 6 μ second load, add and store cycle with 128K, 32 bit words of main memory. The major standard peripherals include line printers—a high speed and medium speed printer; mass storage devices—a rapid access fixed-head disc, medium capacity dual controller disc, two high capacity moving-head discs, and four 9-track tape drives; and terminals—three CRT/keyboards, eight teleprinter/keyboards, two low-speed dial-up modems, and four medium speed RS-232C data links.

6.4.1.2 Operations Control Center

The OCC, Figure 6-14, provides the focal point for EOS multi-vehicle mission operations. Based on the payload scheduling and orbital data information received from the DME, it

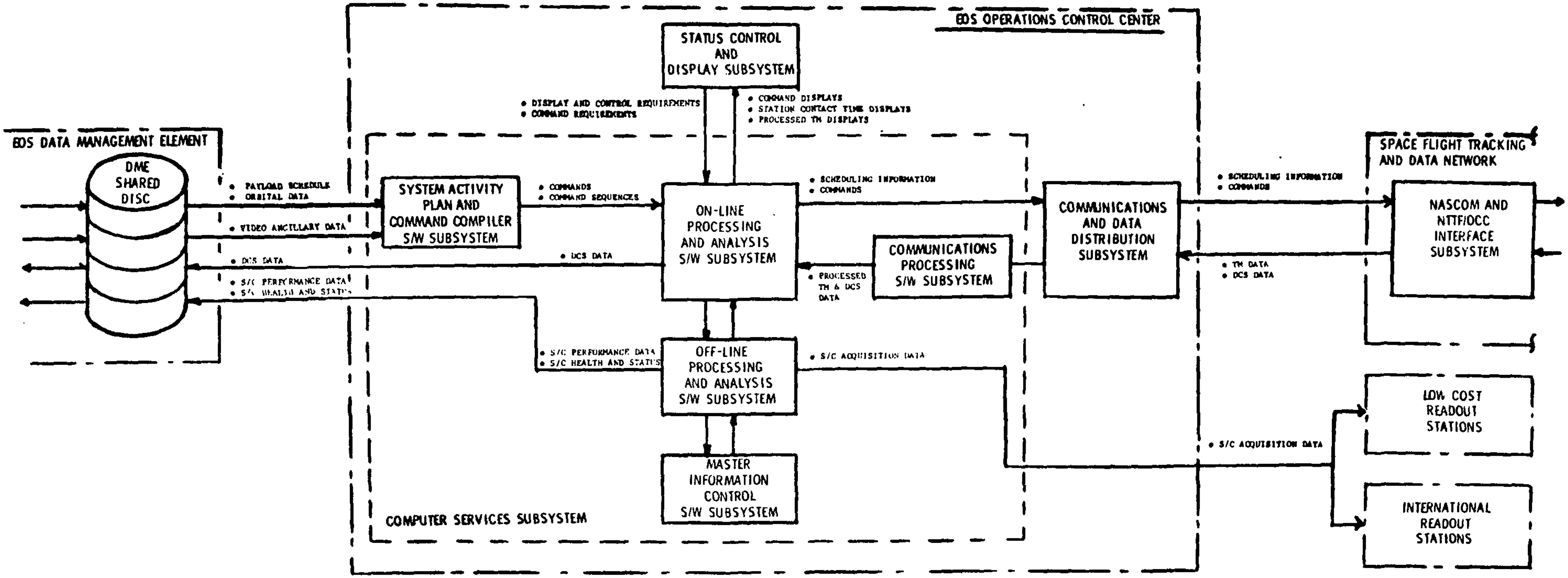


Figure 6-14. Operations Control Center Functional Block Diagram

provides the integrated system activity plan for the instruments, spacecraft and network scheduling, and generates the necessary commands which are transferred to the spacecraft via the Spaceflight Tracking and Data Network (STDN). Upon receipt of spacecraft telemetry data via STDN, the OCC processes the data, and provides spacecraft performance and status information to the DME. The OCC also strips out the DCS data and provides the DME with valid reformatted DCS data for use in the generation of DCS products. The exchange of data between the OCC and DME is through a 24 Megabit shared disc. In addition, the OCC provides spacecraft acquisition information for use by the Low Cost Readout and International Stations.

The OCC is required to support multi-spacecraft operations with the instrument scheduling being performed by the DME, and the complete spacecraft systems activity plan, which incorporates the payload scheduling inputs, generated within the OCC.

The results of the OCC/DME computer configuration trade-off study identified the preferred configuration to be three medium scale computers (one for the DME and two for the OCC) with a 24 Megabit shared disc for data exchange between the OCC and DME to minimize manual data transfer. The OCC computers and peripheral equipment are configured into a separate on-line operational configuration and off-line analysis configuration with switching equipment to provide full back-up capability in event of a failure.

Utilization of the on-board computer provides an extension of the OCC for real-time spacecraft management functions such as power management, antenna pointing, ancillary data insertion, limit checking, alarm and correction.

The OCC hardware design approach utilizes new equipment for the computer services subsystem with improved ERTS like designs for the status, control and display subsystem, and communications and data distribution subsystem. The OCC software approach utilizes the basic ERTS software subsystem designs modified/improved to support EOS multi-vehicle mission operations and the all new computer configuration. The resulting OCC design approach is compatible with the standard EOS spacecraft with flexibility to adapt

to future missions primarily through minor software changes (those associated with changes of mission peculiar payload operation).

6.4.1.3 Image Processing Element

The IPE provides the capability of processing the Thematic Mapper (TM) and High Resolution Pointable Imager (HRPI) instrument data recorded at the remote sites on video tapes. It is responsible for screening, processing, and correcting all of the raw instrument data and generating high density digital tapes of the corrected instrument data for archiving purposes. It is also responsible for generating standard and custom products as directed by the DME by means of work orders and providing production station product definition and output products to the DME for overall management control, monitoring, and reporting.

The IPE design approach, as shown in Figure 6-15, is configured to perform standard on-line processing functions for all the valid TM and HRPI imagery data contained on the raw video input tapes, generate high density digital output tapes of the corrected imagery data and to perform custom off-line processing to generate user custom products on selected data contained on the corrected HDDT's.

The on-line portion of the IPE performs all the standard radiometric and geometric correction functions in the initial processing of the TM and HRPI data. These functions are performed utilizing ancillary data (image processing data transmitted to the spacecraft from the ground and real-time attitude control data from the spacecraft) inserted into the instrument video data stream.

The on-line portion of the IPE is a two-pass system. During the first pass, performed at real-time rates, the raw video data received from the spacecraft (equivalent to 230 scenes/sensor/day) is evaluated for quality and cloud coverage, and descriptive catalog files are generated. Preliminary radiometric correction required to facilitate quality screening and evaluation of Ground Control Point (GCP) image data is also performed. In addition, correction data required for the second pass for full radiometric and X and Y geometric correction are generated.

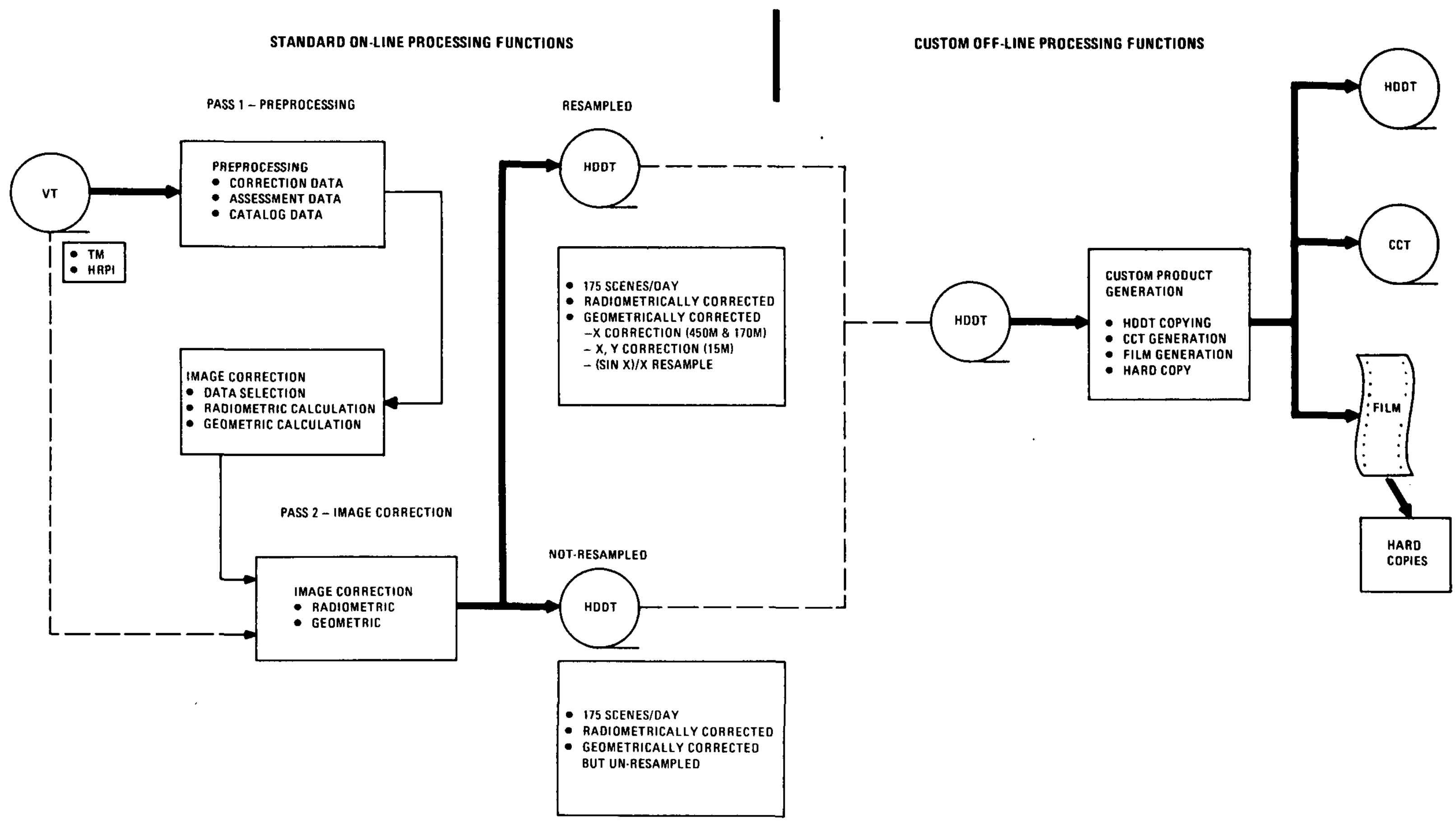


Figure 6-15. Image Processing Element

Radiometric corrections and geometric corrections are performed on the data during the second pass. Image correction of the data is more costly and slower to perform than the preprocessing first pass; hence, throughput is maximized by elimination of unusable data and tape gaps identified in the first pass. All U.S. data (equivalent to 40 scenes/sensor/day) is radiometrically corrected to ± 1.6% and geometrically corrected in the X and Y direction using ground control points to achieve ± 15 meter accuracy on a space oblique mercator projection utilizing cubic $\frac{\sin x}{x}$ resampling. All non-U.S. data (equivalent to 135 scenes/sensor/day) is radiometrically corrected to ± 1.6% and geometrically corrected in the X direction only to achieve ± 450 meter accuracy (using predicted ephemeris) or ± 170 meter accuracy (using best fit ephemeris) on a best fit planar projection utilizing cubic $\frac{\sin x}{x}$ resampling. Outputs of the second pass are two HDDT's—one containing radiometrically corrected resampled data, the second containing radiometrically corrected non-resampled data with geometric correction information included on the tape. This second tape is for use in generating special custom products requiring nearest neighbor or bilinear resampling.

Two viable implementation approaches are available for performing the standard on-line processing function. The <u>micro-programmable processor</u> approach has the advantage of flexibility (through software changes) and a minimum of new equipment design; the <u>special purpose processor</u> approach has a cost advantage for a specified set of requirements.

The custom off-line portion of the IPE can be subdivided into three independent functional areas: digital tape generation and copying, film image generation and processing, and extractive processing/browse file access, all of which can be executed simultaneously.

The design of the digital tape generation and copying subsystem is based on the maximum utilization of equipment (primarily recorders) to meet throughput requirements to minimize total cost. The HDDT subsystem is designed to produce multiple copies (10 copies of U.S. data and 4 copies of non-U.S. data). The CCT generation subsystem is designed to perform custom processing functions (35 scenes/day) and produce CCT's as its normal output; the CCT subsystem is also capable of outputting to an HDDT the same custom

processed data for hard copy reproduction in the film image generation and processing subsystem.

The design of the film image generation and processing subsystem is based on the generation of intermediate HDDT's from the standard on-line HDDT's, to achieve maximum (near continuous) utilization of the expensive laser beam recorders and minimize total costs. The film image generation subsystem provides the capability for catalog film image generation (175 scenes/day of one selected spectral band from each sensor) on five-inch film format and custom film image generation (60 scenes/day/sensor) on 9.5-inch format. The existing ERTS photo laboratory will provide photocopying of the catalog film, and generation/photocopying of black and white and color images with/without enlargement and prints to satisfy user custom product requirements.

A single design for both extractive processing and browse file access provides a cost effective approach since both functions utilize common control, processing and display hardware. This subsystem permits user interaction to search the archived data (narrative description catalog and catalog film) for availability of suitable data. Examination of requested imagery data from either archived HDDT's or specially generated custom CCT's can be accomplished by remote terminal displays. Hard copy information, such as printouts or hard copy color or black and white prints can be created; in addition, custom CCT's can be generated for processing by the film image generation and processing subsystem for creation of film images and/or prints. The user can be provided, on request, copies of HDDT's, CCT's, film images, and/or prints for further analysis at his own facility.

6.4.2 LOW COST READOUT STATION

The Low Cost Readout Stations (LCRS's) provide the local user the capability to acquire and track the EOS Satellites, receive and record useful sensor image data in real-time over their regional areas of interest, and to process, correct, display, and analyze the image data in a timely manner. The LCRS design concept, Figure 6-16, utilizes a standard front end configuration for the data acquisition, processing, and correction

functions to provide the required basic capabilities to the local users at minimum investment. Extractive processing and data display functions have not been included in the basic capability since they must satisfy the unique requirements of the individual local users.

The design approach for the standard portion of the LCRS utilizes a 1.8 meter antenna (preprogrammed for open-loop tracking by means of punched paper tape), an uncooled low noise amplifier, an FM receiver and discriminator, 20 Mb/s bit synchronizer, and a 14 channel linear multi-track high density digital tape recorder for receiving and recording either the 15 Mb/s geometrically uncorrected full five band Multispectral Scanner (MSS) or one of four selectable modes of on-board geometrically corrected Compacted Thematic Mapper (CTM) image data. The Data Acquisition Subsystem provides the capability to acquire data from the satellites over a regional area defined by a 500 KM radius from the LCRS.

Post pass processing and full radiometric correction of the individual spectral bands of the instrument is accomplished at approximately a 40:1 reduction rate (approximately 20 minutes/scene) utilizing the high density digital tape recorder in a reduced playback mode, a variable bit synchronizer, newly designed demultiplexer and I/O control unit, a mini-computer (characterized by a 1.0 μ second cycle time, 16K memory and a 16-bit word size), associated peripheral equipment, and a 9-track controller and magnetic tape unit. Output of the Data Processing and Correction Subsystem is a standard computer compatible tape (CCT) of the corrected and annotated image data.

Generation of visual image displays and/or photographic image film from the corrected and annotated image data contained on the CCT's is accomplished off-line utilizing the same controller and magnetic tape unit in a playback mode, the mini-computer and peripheral equipment contained within the standard configuration, and the display, recording, and auxillary equipment provided in the station unique Data Display and Extractive Processing Subsystem.

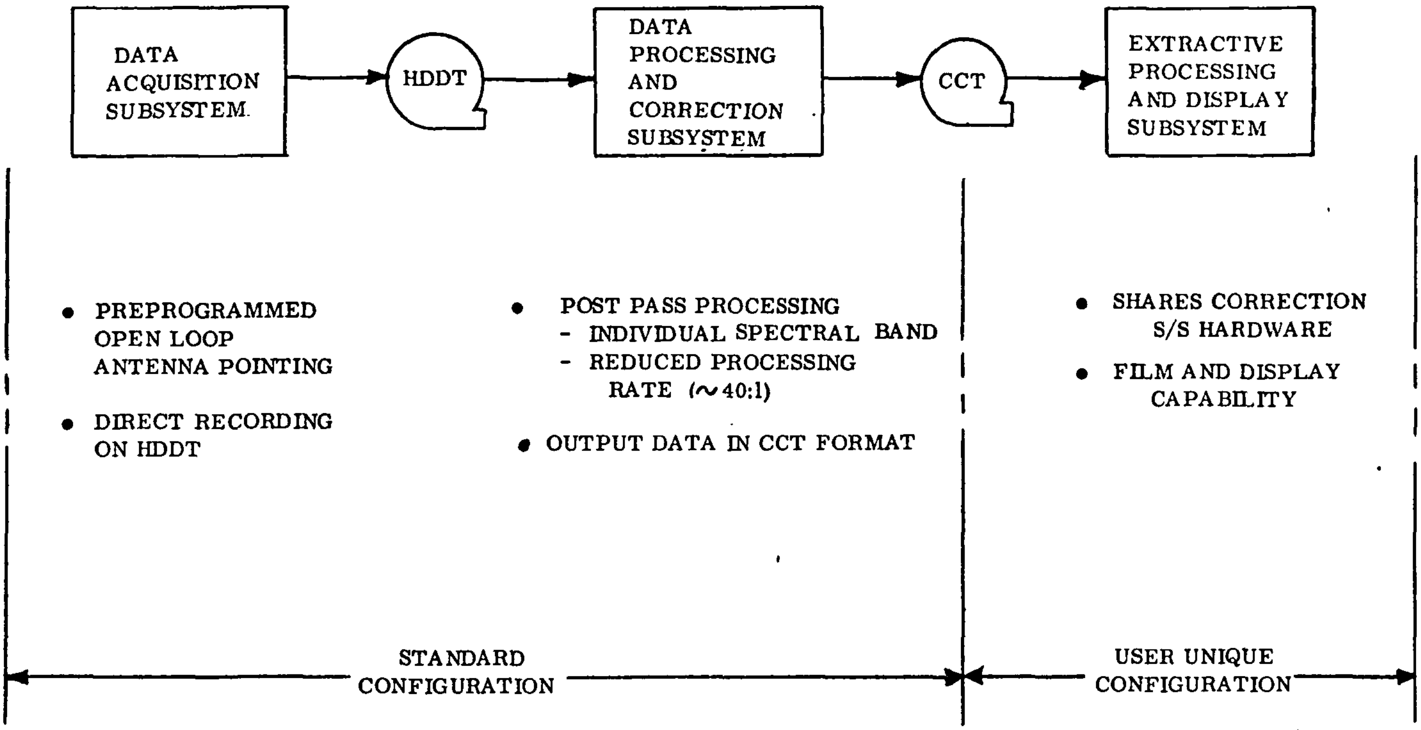


Figure 6-16. Low Cost Ground Station

6.5 SPECIFICATIONS

Specifications for the EOS system have been prepared, as shown in Figure 6-17, for all system elements. For the spacecraft, these specifications are segregated into two parts; those associated with the basic multi-mission spacecraft, and those associated with EOS-A mission peculiar equipment. These specifications are written at the subsystem level. The ground system specifications are written at the major segment level and address (in the OCC and CDPF) the requirements of the EOS-B mission. For convenience, the volume numbers of Report No. 5 in which each of these specifications appear is shown on the figure.

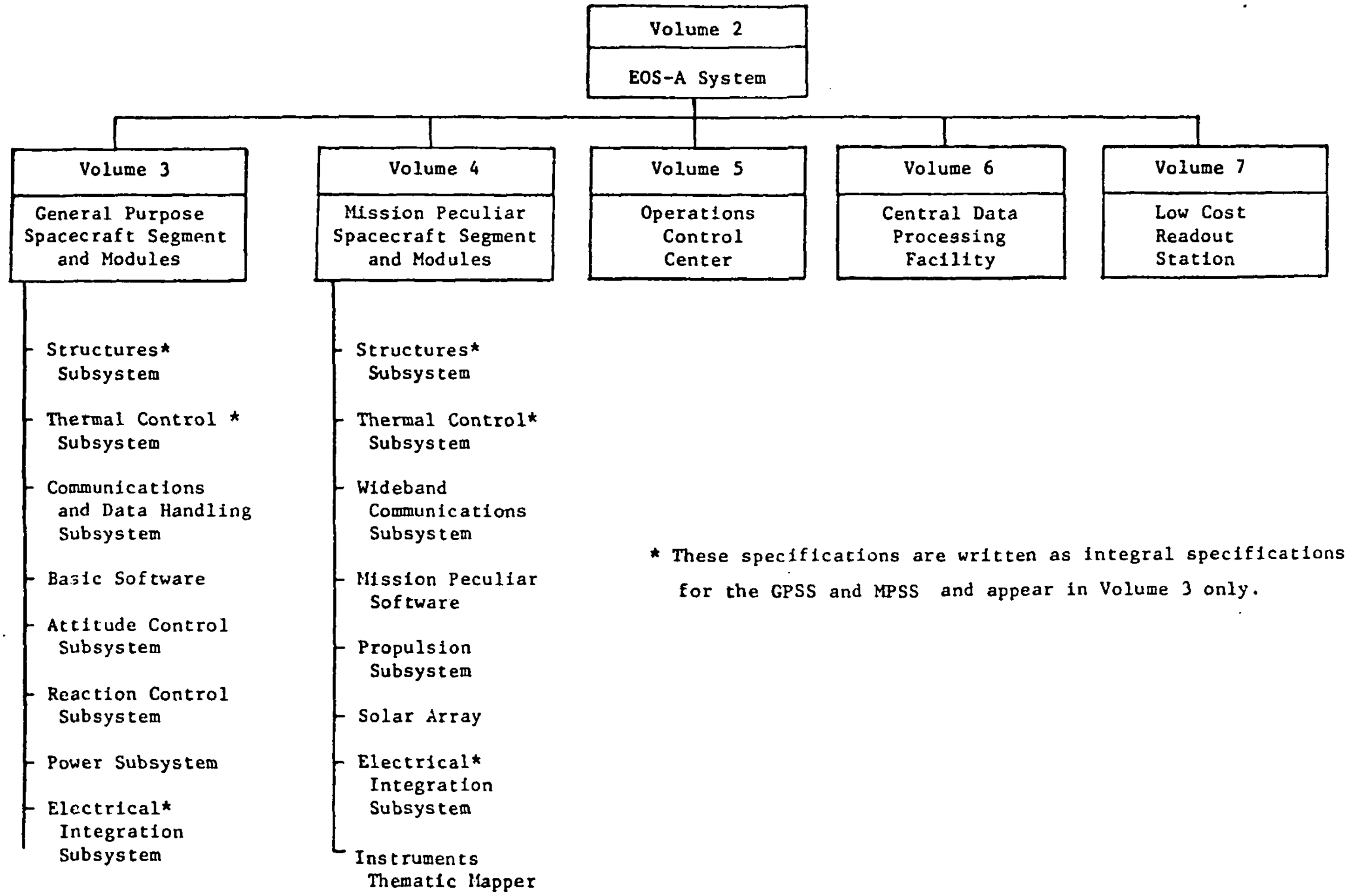


Figure 6-17. EOS Specification Tree

SECTION 7.0

SPACE SHUTTLE INTERFACES/UTILIZATION

EOS is the first spacecraft being designed to be compatible with Space Shuttle. The investigations of the mechanical and electrical interfaces, the impact of Shuttle operations, and the cost benefits accrued from Shuttle utilization therefore become key areas of interest in the EOS study. This section documents these investigations and quantifies the significant cost savings which result.

Since the design definition of Shuttle is evolving and being compiled in preliminary form, flexible implementation concepts for EOS are required. This is especially true in the electrical interface area where very little detail Shuttle information is available. Additional Shuttle interface areas requiring better definition are safety criteria, contamination, and environmental control.

7.1 STRUCTURAL/MECHANICAL INTERFACES

The Mechanical Interfaces between EOS and Shuttle have had considerable emphasis since separate studies have been performed by RI and SPAR/DSMA to evaluate a Shuttle bay support system for EOS. The support system consists of:

- Storage fixture
- Launch/retrieval support cradle
- Docking frame and erection mechanism
- SPMS exchange mechanism
- Module storage magazine
- Shuttle attached manipulator system.

This full complement of mechanical support equipment is used during a combined EOS delivery/retrieval and service mission. Reduced sets of this equipment can be used for delivery or retrieve only missions. General Electric has used this hardware definition in establishing the EOS mechanical interfaces with the Shuttle.

The mechanical interfaces and provisions required for a launch or retrieval EOS mission are summarized in Figure 7-1. These features include mechanical interfaces for support of EOS in the Shuttle bay during Shuttle flight and provisions for retrieval or release of EOS by Shuttle. Other features required in the EOS design are provisions to refold appendages such as solar arrays, antennas, and booms, in addition to providing covers for critical equipment. The Shuttle flight support equipment required for this mode of operation consists of the:

- Launch/retrieve support

  (The design of this support has been simplified and the weight reduced by replacing the EOS transition ring with a three-point transition frame.)

- Docking frame and erection mechanism

  (The use of this equipment for the launch and retrieve mission is optional; SAMS may be used to place the spacecraft directly into the launch/retrieve support.)

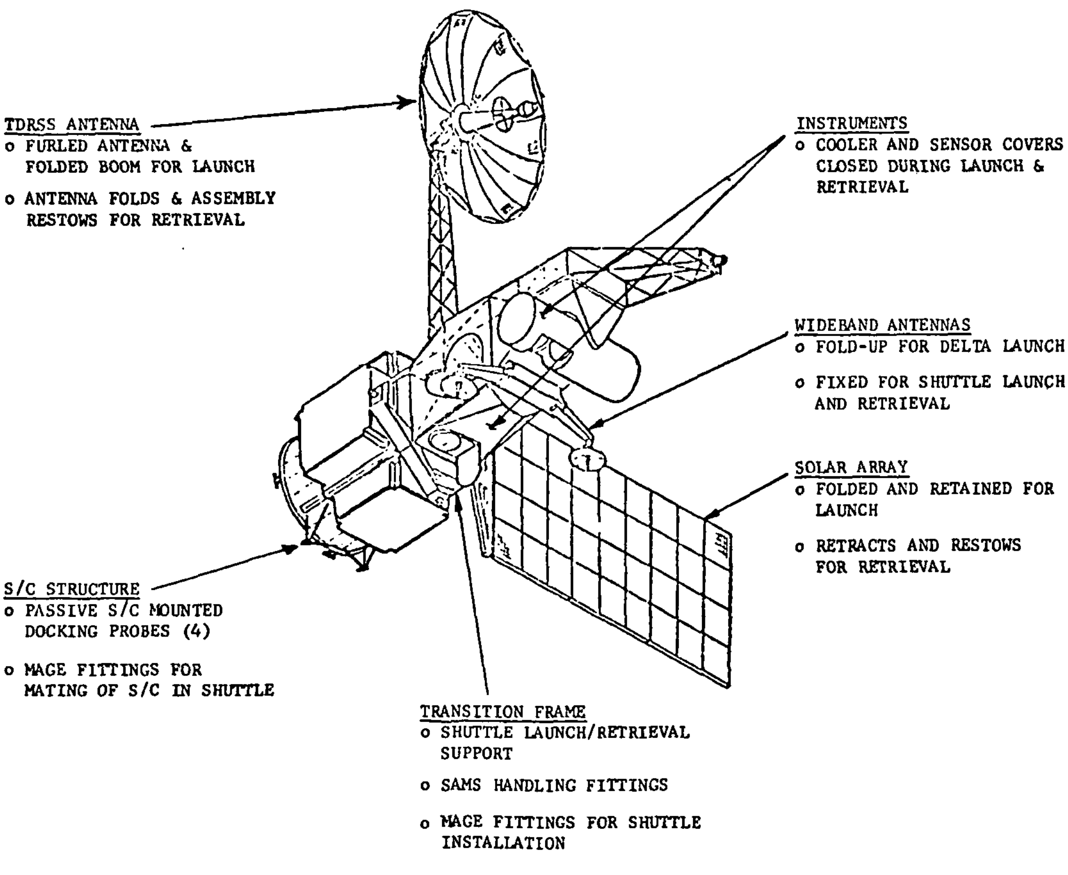


Figure 7-1. EOS Spacecraft Launch/Retrieve Provisions

The mechanical provisions and interfaces required for a resupplyable EOS spacecraft are summarized in Figure 7-2. Features that must be added for the resupply mode include corner latches and remote connectors for the exchange of modules in the shuttle bay. Modularization of the instruments are also required to facilitate this remote exchange. An alteration to the Shuttle attached docking frame and erection mechanism is required to allow axial removal and replacement of the propulsion module. This requirement has been coordinated with R.I. and deemed acceptable.

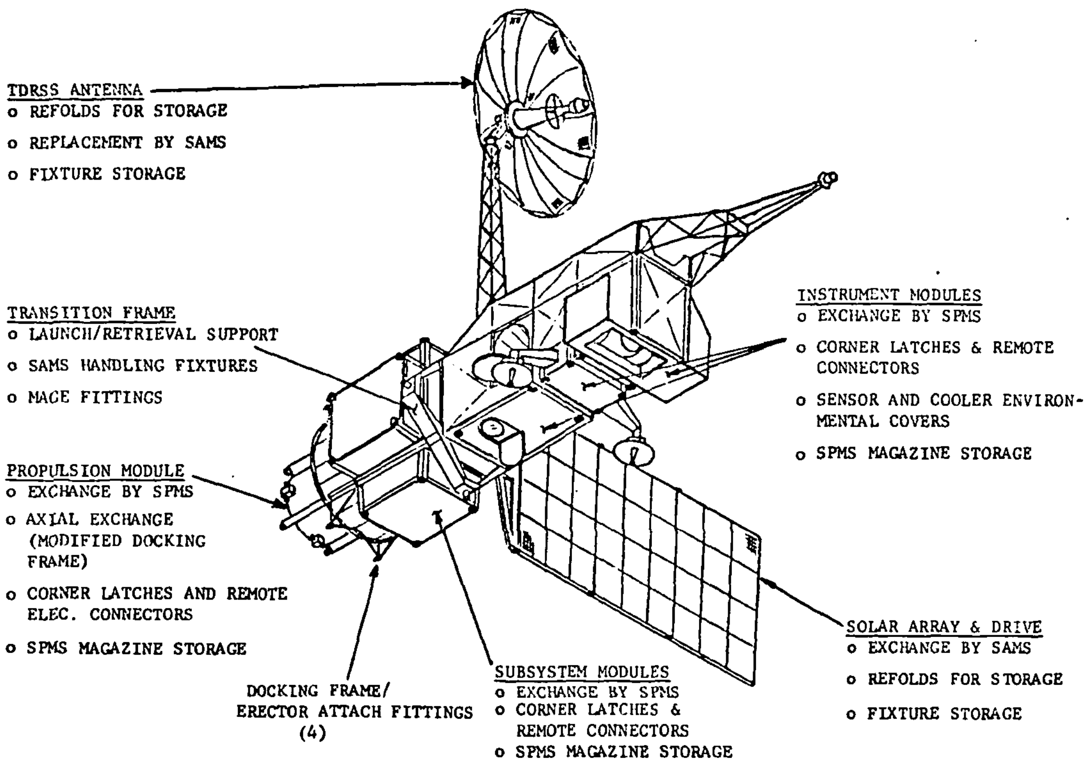


Figure 7-2. EOS Spacecraft Resupply Provisions

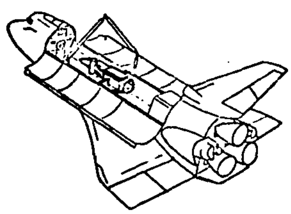
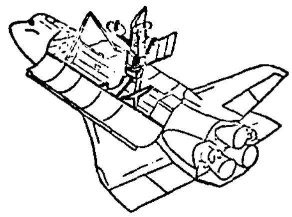
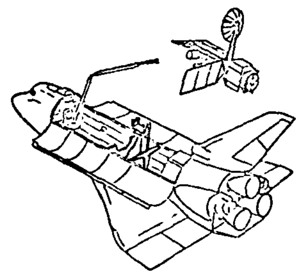
7.2 ELECTRICAL INTERFACES

The Electrical Interfaces between the EOS and Shuttle Orbiter occur under three basic modes of operation:

1. Stowed in the Shuttle bay and attached to the launch/retrieve support.
2. Attached to the docking frame and erected in a vertical position for module exchange.
3. Detached from Shuttle with Shuttle in a loiter mode.

The attached modes provide hardwire connections for all EOS input and output signals: power, command, telemetry, and caution and warning. The detached mode employs RF communications between the spacecraft and the orbiter, with the spacecraft on-orbit and providing its own power. The detached mode is used to provide a check of the entire spacecraft while the Orbiter is on-station. The electrical interfaces for the three modes are summarized in Table 7-1.

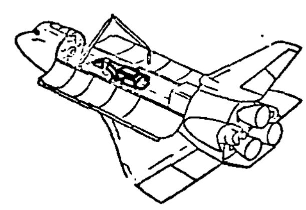
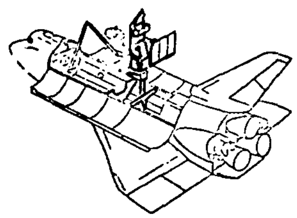
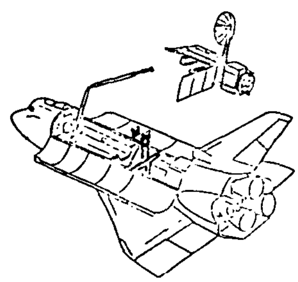
Table 7-1. Summary of Electrical Interfaces

			
INTERFACE	STOWED (HARDWIRE)	POSITIONED (HARDWIRE)	DETACHED (RF)
POWER	REGULATION	REGULATION	EOS INTERNAL
COMMAND	8 Kbps (2Kbps EOS DATA) OBC DECOM	8Kbps (2Kbps EOS DATA) OBC DECOM	PRIME - STDN OR TDRSS BACKUP - SHUTTLE 8 Kbps
TELEMETRY	16 Kbps OBC FORMAT (TBD)	16 Kbps OBC FORMAT (TBD)	PRIME - STDN OR TDRSS BACKUP-SHUTTLE 16 Kbps
DATA	MOD. PROC. 1.024 Mbps MB LCU LINK 50 Mbps WB	MOD. PROC. 1.024 Mbps MB LCU LINK - 50 Mbps WB	PRIME - STDN OR TDRSS/FULL CAPABILITY BACKUP-NONE
CAUTION&WARNING	SHUTTLE DISPLAY (35)	SHUTTLE DISPLAY (35)	NONE

7.3 SHUTTLE OPERATIONS

The Space Shuttle has generally been found to be effective in supporting the Operation of EOS during all phases of the mission. However, in some cases, notably during pre-launch activities, the present mode of operation requires modification to fit the overriding Shuttle operational flow. The final close-out of the payload at approximately T-69 hours is critical since it restricts access to the spacecraft for almost three days prior to launch. Key operational advantages inherent in Shuttle utilization are the added capability of on-orbit checkout which is summarized in Table 7-2 in addition to the capability of retrieving the spacecraft for ground or on-orbit resupply.

Table 7-2. Spacecraft Checkout in Shuttle Orbit

		
SPACECRAFT IN SHUTTLE RETENTION CRADLE	SPACECRAFT ATTACHED TO POSITIONING PLATFORM	SPACECRAFT IN SHUTTLE LOITER MODE
• CAUTION & WARNING MONITORING • STATUS/LIMIT CHECKING OF SUBSYSTEMS & INSTRUMENTS • SPACECRAFT OBP MEMORY UPDATING • PRE-DEPLOYMENT CHECKOUT - HARDWIRE & MECHANICAL INTERFACES - ELECTRICAL CONTINUITY - VISUAL INSPECTION	• CAUTION AND WARNING MONITORING • DEPLOYMENT OF APPENDAGES • STATUS/LIMIT CHECKING AND PRELIMINARY FUNCTIONAL CHECKING OF SUBSYSTEMS AND INSTRUMENTS • SPACECRAFT OBP MEMORY UPDATING • PRE-SEPARATION CHECKOUT - R.F. INTERFACE - ELECTRICAL CONTINUITY - VISUAL INSPECTION • VERIFY RECAPTURE & RETRIEVAL CAPABILITY	• EOS ON SPACECRAFT ACS, POWER AND COMMUNICATIONS • ACTIVATION AND CHECKOUT UNDER GROUND CONTROL • REMAINDER OF SPACECRAFT DEPLOYMENTS (IF REQ'D)

7.4 SHUTTLE MODE COST ANALYSIS

The Cost Benefits accrued from Shuttle utilization is the key study addressed in this section. The cost benefits depend, to a great extent, on how Shuttle is used, be it solely as a low-cost launch vehicle or as an integral part of the total EOS system accomplishing on-orbit resupply to extend the spacecraft life in orbit. Many variables are involved in evaluating the impacts of alternate Shuttle utilizations on the EOS program costs. The approach taken was to assume reasonable values for these variables and to also investigate some key variables (such as refurbishment costs, launch cost, and number of spacecraft failures) parametrically. These parametric analyses allowed greater insight into which variables most impacted the analysis results. Once this insight was gained, program costs were determined for nominal (best estimate) values of the variables and maximum (worst case) values of the variables. These program costs were determined for a nominal mission model having two spacecraft in orbit at one time over a ten-year lifetime for expendable spacecraft, ground serviceable and on-orbit serviceable spacecraft cases.

The results of this analysis are summarized in Table 7-3. In all cases, the on-orbit serviced spacecraft (10-year lifetime) proved lowest cost and the expendable spacecraft proved highest cost. The sole difference between the Option #1 and #2 under the nominal cost case was the method of charging for spacecraft costs. In Option #1, total spacecraft costs for the number of spacecraft required to perform the mission model were charged for a 10-year period independent of the program lifetime that could be expected from these spacecraft. In Option #2, the spacecraft costs were prorated for a ten-year period of a longer program. (For example, if three spacecraft are required to perform the mission model but these spacecraft would last 15 years with refurbishment, the prorated cost for a ten-year program would be only two spacecraft.) For each of the nominal cases considered there is very little difference in cost between the ground serviced spacecraft and the combined ground and on-orbit serviced spacecraft. When the "worst case" variables are considered, the combined ground and on-orbit serviced spacecraft show a decided advantage over the ground serviced spacecraft.

Table 7-3. Shuttle Mode Cost Analysis Summary

CASE	NOMINAL COST M$				ALTERNATE COSTS M$ (Max. Variables)	
	OPTION #1		OPTION #2			
	• Nominal Refurb. • Two Failures • Total S/C Costs • Non-recurring Service Costs	Normalized Cost	• Nominal Refurb. • Two Failures • Prorated S/C Costs • Non-recurring Service Costs	Normalized Cost	• High Refurb. • Three Failures • Total S/C Costs • Mod High Launch Costs • Non-recurring Service Costs	Normalized Cost
Expendable Spacecraft (Single Launch)	382	1.90	382	1.90	459	1.71
Expendable Spacecraft (Dual Launch)	382	1.90	382	1.90	428	1.59
Ground Service Spacecraft (Single Launch)	236	1.17	212	1.05	381	1.42
Ground Service Spacecraft (Dual Launch)	258	1.28	213	1.06	352	1.31
Ground and On-orbit Serv S/C (One Service and Return)	236	1.17	215	1.07	306	1.14
Ground and On-orbit Serv S/C (Two Service and Return)	233	1.16	213	1.06	300	1.12
Orbit Service Spacecraft (Six-year Life)	257	1.28	237	1.18	317	1.18
Orbit Service Spacecraft (Ten-year Life)	201	1.00	201	1.00	269	1.00

The following recommendations can be made from the cost analysis:

- The most cost-effective use of Shuttle is achieved by using it to deliver the spacecraft and also assist in servicing the spacecraft to extend its orbital lifetime.
- The Shuttle launched EOS spacecraft should be designed for on-orbit servicing; the spacecraft launched prior to Shuttle availability can be designed for Shuttle retrieval and ground servicing without incurring significant cost penalties over on-orbit servicing.
- As designs of EOS and Shuttle mature, the Shuttle analysis can be refined to establish the most cost-effective use of Shuttle and the optimum interval for Shuttle service. This may include combined on-orbit and ground servicing or may be limited to on-orbit servicing of the spacecraft.

7.5 SHUTTLE ORBIT TRADES

The Shuttle Orbit Trades indicate that it is cost effective to include orbit transfer capability on-board the EOS spacecraft allowing Shuttle delivery and retrieval at an altitude range between 465 and 610 km (250 and 330 nm) independent of EOS mission altitude (see Figure 7-3). The mission impacts of servicing the spacecraft at a low altitude (compatible with Shuttle high payload capability) and returning to the mission orbit of 775 km (418 nm) have been investigated without uncovering any significant problems. This concept of low altitude servicing allows consideration of servicing multiple spacecraft as long as the spacecraft are in approximately the same orbital plane. The mission altitude selected also permits direct Shuttle access (at higher cost) in the event of a spacecraft failure which prevents its return to the service altitude.

7.6 SAFETY CONSIDERATIONS

The Safety Considerations on EOS differ from previous automated payload requirements in that these programs were primarily concerned with safety during the ground flow while the use of Shuttle requires that this concern be extended into the flight phases of the mission. This added set of requirements potentially impacts all aspects of the system design and development. The results of a safety review indicate, however, that only relatively minor design modifications are required to "safe" an EOS spacecraft for a Shuttle launch.

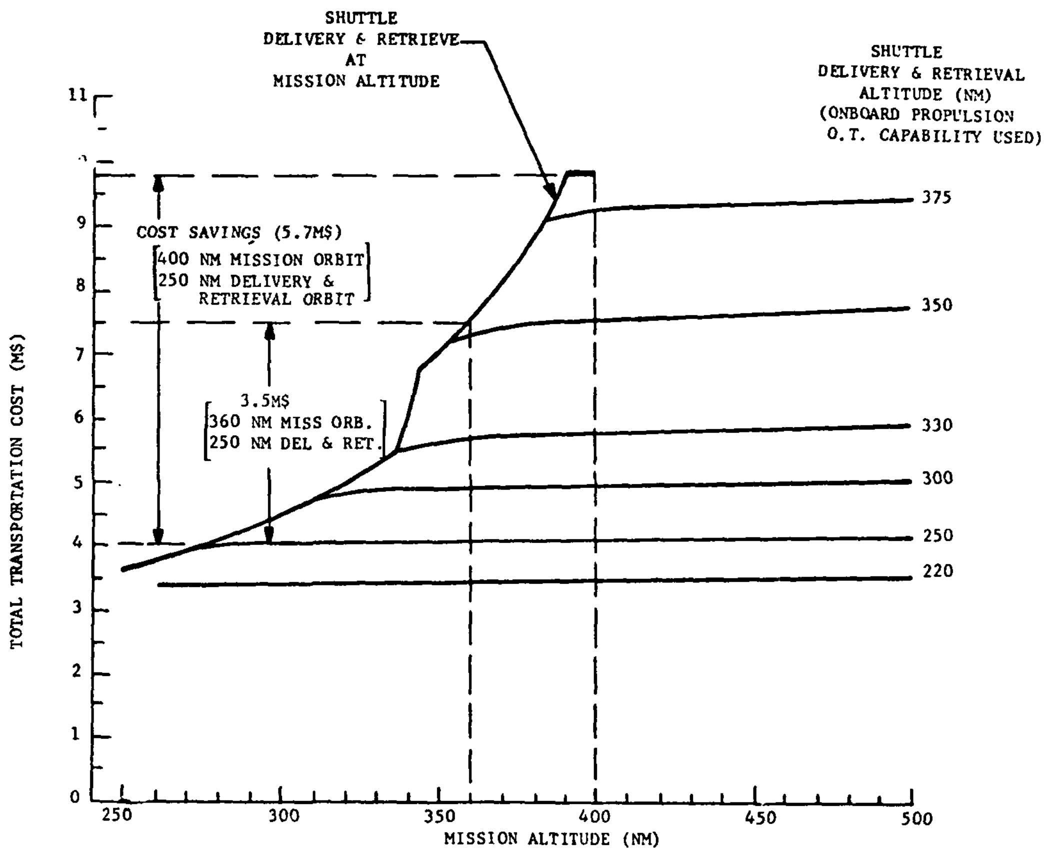


Figure 7-3.
Transportation Cost Savings with Shuttle Delivery and Retrieval at Low Altitudes

7.7 CONTAMINATION CONTROL

A preliminary investigation was conducted of the compatibility of EOS and the Shuttle-induced environment. Contamination and Thermal Control were the two major areas of concern. With respect to Contamination Control it was felt that the control of gaseous and particulate contamination does present a potential problem to the optical and radiative cooler surfaces of the EOS instruments. Several design and operational countermeasures have been developed for both the Shuttle and the EOS. The general conclusion of the analysis is that methods and techniques do exist which can successfully cope with potential contamination problems.

7.8 THERMAL CONTROL

The analysis of Shuttle Thermal Control provisions was conducted with the currently available Shuttle cargo bay thermal environment data. The results indicate that the spacecraft can be integrated with the Shuttle by taking the following steps:

1. Limit the maximum ground conditioning air temperature to 86° F to ensure that the battery temperatures are maintained below their maximum allowable transient temperature of 95° F during launch.
2. Select spacecraft orientations in the Shuttle bay to ensure that critical components (such as batteries) are located away from local "hot spots" that occur in the payload bay during re-entry.

SECTION 8.0

CONCLUSIONS AND RECOMMENDATIONS

Based upon the myriad design/cost trade studies performed during the EOS Definition Study Program and those completed previously, the Space Division of the General Electric Company is convinced that the Aerospace Team can effectively lower the overall cost of space developments. The standardization concept and the repeated use of a flexible modularized basic spacecraft offer a clear-cut way to eliminate much of the development costs for succeeding users. The launch, retrieve, and on-orbit service capability of the space shuttle affords an even more dramatic opportunity to reduce the cost of both operational and R&D space programs. EOS, because of the many missions (see Figure 8-1) involved, is a logical program to initiate a concerted effort to design, develop, and manufacture a series of standard basic spacecraft which can be effectively utilized to provide the vehicle by which many developmental/operational payloads can be carried into earth orbit.

The salient conclusions and recommendations derived from the EOS Definition study are:

- The standard general purpose spacecraft concept is technically sound and is cost effective across multiple missions.
- Modularity at the subsystems level simplifies spacecraft interfaces and spacecraft integration and test.
- The modular spacecraft design used initially with conventional launch vehicles can be made compatible with shuttle with minor impact.
- The standard modular spacecraft can be designed for launch on conventional boosters, such as Delta and Titan, and then transition into the shuttle era without redesign.
- The standard modular spacecraft can be built and tested for a recurring cost of about seven million dollars.
- Combination of both R&D and operational mission payloads using a common spacecraft is cost effective in the earth resources missions examined.

- Considerable reductions in program cost can be achieved by using shuttle for both launch and spacecraft service.
- Pre-shuttle launched spacecraft should be designed for retrieval by shuttle.
- Shuttle launched spacecraft should be designed for on-orbit servicing.
- A state-of-the-art ground system can be built to process wideband data from instruments such as the Thematic Mapper and High Resolution Pointable Imager at costs comparable to those for ERTS but capable of processing an order-of-magnitude more data. Special purpose hardware or microprocessors (rather than general purpose computers) are required to achieve this capability.
- Local stations to receive and process limited, but useful, data directly from the spacecraft can be built for a recurring cost of about $200,000.
- Traditional concepts of extensive testing at all levels of assembly can be relaxed for a modular spacecraft resulting in significant cost savings. This is particularly true in the era of on-orbit repair and retrieval since most malfunctions lose their catastrophic implications.

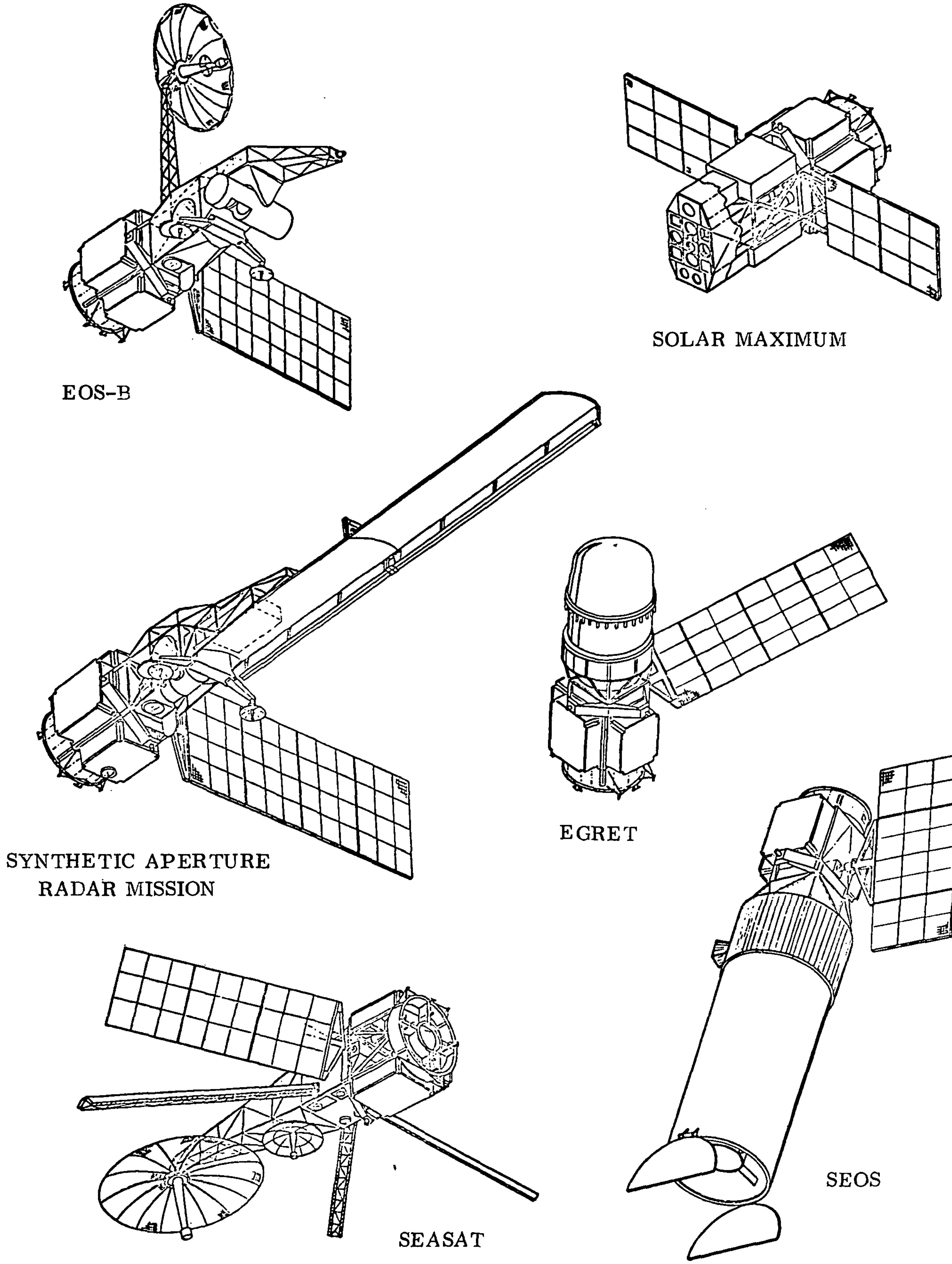


Figure 8-1. EOS Alternate Mission Configurations

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